NASA CR71513

VOYAGER SPACECRAFT SYSTEM

FINAL TECHNICAL REPORT

TASK B

VOLUME A

PREFERRED DESIGN FOR FLIGHT SPACECRAFT AND HARDWARE SUBSYSTEMS

PART I

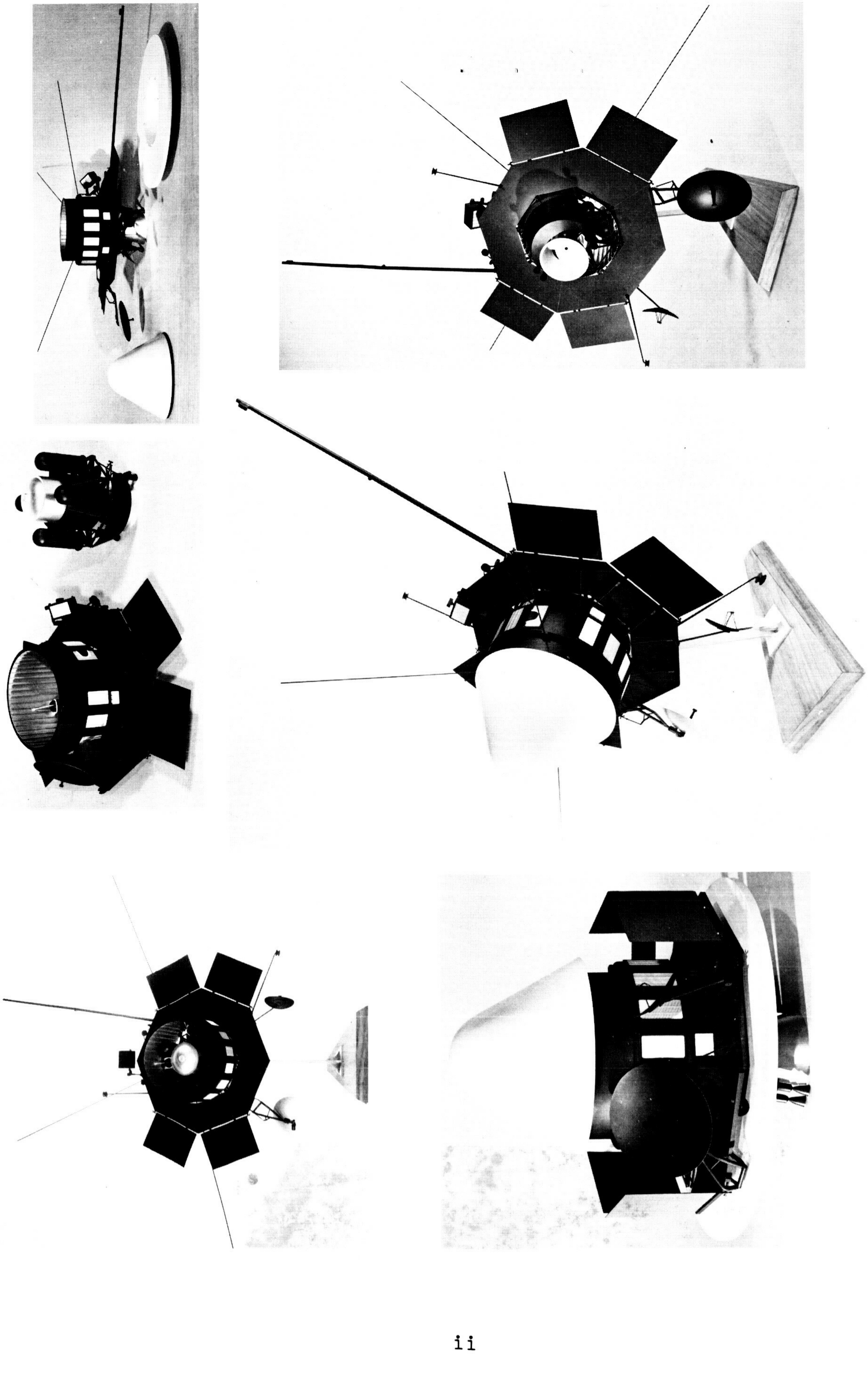
D2-82709-6

prepared for

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

UNDER
CONTRACT NO. 951111 -NAS-7100
JANUARY 1966

THE BOEING COMPANY • SPACE DIVISION • SEATTLE, WASHINGTON



Voyager Spacecraft 1/10 Scale Mockup

THE BOEING COMPANY

AEROSPACE GROUP • P. O. BOX 3707 • SEATTLE, WASHINGTON 98124

January 21, 1966

IN REPLY REFER TO

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena, California

Gentlemen:

The Boeing Company is pleased to submit the technical reports of the work accomplished under Voyager Phase 1A, Task B. Together with the reports of Task A, they represent to us a substantial contribution to our understanding of the objectives of the Voyager Project. As a corollary, it is believed they will demonstrate to you a dedication for, and a capability to perform, those tasks so important to fulfilling the Spacecraft Contractor's responsibilities.

The recently announced delay in the Voyager Program will test the dedication of all parties concerned. Despite our disappointment, we will not let this temporary setback deter our proceeding on a rational basis to be ready when funding levels again allow the program to proceed. It is important to note that the Task B documentation has been submitted as if no change had occurred in the Voyager Program. It should be recognized that corporate and group commitments contained in the documentation, in the areas of facilities and personnel, will be reconsidered when the Voyager program proceeds. At that time, Boeing will update and reaffirm the resources necessary to support the Voyager program.

Because of the cancellation of the Phase 1B, Part 2 Request for Proposal, we have chosen to highlight some of our management philosophy and organization rationale in a summary document, D2-82709-00. To place this in perspective, we have also summarized the salient features of the spacecraft design. Further, we have postulated some advanced missions, using the 1971 design, for further exploration of the solar system. This latter item is the basis for part of our continuing Voyager work.

Little more remains to be said except to restate that the Voyager Spacecraft System represents to us, more than a new product objective; it is an opportunity to participate in the extension of scientific knowledge in the universe and to contribute to national prestige.

Very truly yours,

THE BOEING COMPANY

Lysle A Wood

Lysle A. Wood
Group Vice President-Aerospace

CONTENTS

		PAGE
INTRODUCTION		I-1
1.0	VOYAGER 1971 MISSION OBJECTIVES AND DESIGN CRITERIA	1-1
1.1	Program Objectives	1-1
1.2	1971 Mission Objectives	1-2
1.3	Mission Constraints	1-3
1.4	Design Criteria	1-4
1.5	System Elements and Weight Allocation	1-4
1.6	Competing Characteristics	1-7
1.7	Mission Trade Study Summaries	1-9
2.0	DESIGN CHARACTERISTICS AND RESTRAINTS	2-1
2.1	Design Characteristics	2-1
2.1.1	Mission Profile	2-3
2.1.2	Science Subsystem Characteristics and Requirements	2-8
2.1.3	Flight Spacecraft Subsystems	2-11
2.2	Design Restraints	2-21
2.2.1	Radiation Sources	2-21
2.2.2	Magnetic Interference	2-22
2.2.3	Environment	2-23
2.2.4	Attitude Control Restraints	2-35
2.2.5	Flight Perturbation Restraints	2-37
2.2.6	Restraints on Separation of Flight Spacecraft and Flight Capsule	2-37
2.2.7	Reliability	2-37
2.2.8	Safety	2-38
2.3	Guidance and Navigation--Maneuver Accuracy and Propulsion Requirements	2-39
2.3.1	System Velocity Control Accuracy Requirements	2-39
2.3.2	Trans-Mars Trajectory Accuracy Requirements	2-39
2.3.3	Trajectory Maneuver ΔV Requirements	2-40
2.3.4	Orbit-Insertion Accuracy Requirements	2-40
2.3.5	Orbit-Insertion ΔV Requirements	2-41
2.3.6	Orbit-Trim Accuracy Requirements	2-41
2.3.7	Orbit-Trim ΔV Requirements	2-42
2.3.8	Mars-Entry Accuracy Requirements	2-42
2.3.9	Mars-Entry ΔV Requirements	2-42
2.4	Aiming--Point Selection	2-43
2.4.1	Aiming--Point Objectives	2-44
2.4.2	Selected Aiming Points for an Example Mission	2-45
2.5	Planetary Quarantine Requirements	2-47
2.5.1	Summary	2-47
2.5.2	Planetary Quarantine Allocation of Event Probabilities	2-49
2.5.3	System Impact Probability	2-51

CONTENTS
(Cont.)

	Page
2.5.4 Meteroid Impact Ejecta Considerations	2-52
2.5.5 Propulsion and Reaction Control Subsystems	2-61
2.5.6 Conclusions	2-67
3.0 SYSTEM-LEVEL FUNCTIONAL DESCRIPTION	3-1
3.1 Standard Trajectories	3-1
3.1.1 Transit Trajectories	3-1
3.1.2 Arrival Date Separation Considerations	3-11
3.1.3 Mars Orbit	3-16
3.1.4 Flight Capsule Entry Trajectory	3-39
3.2 Orbit/Entry Determination Uncertainties	3-41
3.2.1 Initial Mars Orbit Determination	3-41
3.2.2 Flight Capsule Entry and Impact Conditions	3-42
3.3 Voyager Flight Spacecraft Components Design Parameters	3-43
3.4 Voyager Equipment Elements and Documentation Identification	3-49
3.5 System Interface Definition	3-49
3.5.1 Interface of Spacecraft System and Capsule System	3-49
3.5.2 Interface of Spacecraft System and Launch Vehicle System	3-57
3.5.3 Interface of Spacecraft System and Tracking and Data System	3-57
3.5.4 Interface of Tracking and Data System and Mission Operations System	3-58
3.5.5 Interfaces of Spacecraft System and Mission Operations System	3-58
3.5.6 Interface of Capsule System and Mission Operations System	3-58
3.6 Telemetry Criteria	3-59
3.7 Telemetry Channel List	3-63
3.8 Guidance and Navigation Maneuver Error Allocations	3-63
3.8.1 Pointing--Angle Accuracy	3-65
3.8.2 ΔV Magnitude Accuracy	3-65
3.9 Flight Sequence	3-65
3.9.1 Pre-launch	3-68
3.9.2 Launch and Injection	3-68
3.9.3 Celestial Reference Acquisition	3-73
3.9.4 Interplanetary Cruise	3-76
3.9.5 Trajectory-Correction Manuevers	3-78
3.9.6 Flight Capsule Canister Separation	3-83
3.9.7 Mars-Orbit Insertion	3-83
3.9.8 Planetary Vehicle Orbital Operations	3-85

CONTENTS
(Cont.)

		Page
3.9.9	Separation of Flight Spacecraft and Flight Capsule	3-88
3.9.10	Capsule Operations	3-88
3.9.11	Flight Spacecraft Orbital Operations	3-89
3.10	Preferred Spacecraft Layout and Configuration	3-89
3.10.1	Summary	3-89
3.10.2	General Description of Preferred Design	3-91
3.10.3	Selection of the Preferred Design	3-106
3.11	Selection of Preferred Spacecraft System	3-119
3.11.1	Summary	3-120
3.11.2	Evaluation Criteria	3-124
3.11.3	Selection Rationale	3-125
3.11.4	Reallocation of Spacecraft System Weights	3-128
3.11.5	System Concept Trade Studies	3-133
3.12	Planetary Quarantine	3-138
3.13	Voyager Flight-Equipment Cleanliness	3-140
3.14	Magnetics	3-141
3.14.1	Magnetic-Control Plan	3-141
3.15	Space-Radiation Effects on Voyager System	3-145
3.15.1	Semiconductor-Device Response	3-146
3.15.2	Voyager-System Materials	3-146
3.16	Data System Description	3-147
3.17	System Reliability Summary	3-155
3.17.1	Reliability Requirements	3-159
3.17.2	Preferred System Evaluation	3-162
3.17.3	Subsystem Evaluation Summary	3-177

INTRODUCTION

This document, D2-82709-6, (Volume A, Part I), "Preferred Design for Flight Spacecraft and Hardware Subsystems" is submitted by The Boeing Company in response to Contract 951111, Phase IA, Task B, dated November 2, 1965. It supplements D2-82709-1 (Volume A, Part I) submitted July 29, 1965, following completion of Phase IA, Task A. The complete technical report in response to Contract 951111, Phase IA, Task B consists of the following:

VOLUME A — D2-82709-6 AND D2-82709-9* — PREFERRED DESIGN FOR FLIGHT SPACECRAFT AND HARDWARE SUBSYSTEMS

PART I

- SECTION 1 - VOYAGER 1971 MISSION OBJECTIVES AND DESIGN CRITERIA
- SECTION 2 - DESIGN CHARACTERISTICS AND RESTRAINTS
- SECTION 3 - SYSTEM LEVEL FUNCTIONAL DESCRIPTION

PART II

- SECTION 4 - FUNCTIONAL DESCRIPTION OF SPACECRAFT HARDWARE SUBSYSTEMS
- SECTION 5 - PROGRAM SCHEDULE AND IMPLEMENTATION PLAN

VOLUME B — D2-82709-7 — DESIGN FOR THE OPERATIONAL SUPPORT EQUIPMENT

VOLUME C — D2-82709-8 AND D2-82709-10* — ALTERNATE DESIGNS CONSIDERED FOR SPACECRAFT PROPULSION SYSTEMS

*CLASSIFIED SUPPLEMENT TO VOLUME A AND C RESPECTIVELY

The highlights of the above documentation and management planning are summarized below.

During the period covered by Contract 951111, Task B, Boeing has revised the preliminary design of the Voyager Spacecraft System in consonance with the statement of work. As part of this effort, Boeing has:

1) Verified and revised the requirements and constraints which are imposed upon the Voyager Spacecraft System by the Voyager 1971 Mission.
2) Reviewed and revised the preliminary Flight Spacecraft design for the Voyager 1971 mission, including the study of alternate designs for the spacecraft propulsion systems.
3) Selected a preferred design which reliably and effectively achieves the objectives of the 1971 mission.
4) Reviewed and revised the functional descriptions for the Flight Spacecraft and for each of its hardware subsystems.
5) Reviewed and revised the preliminary requirements and functional description for the Operational Support Equipment (OSE) necessary to accomplish the 1971 mission.
6) Updated and revised the schedule of the Voyager Implementation Plan.

The Boeing Voyager Spacecraft System organization, shown in Figure I-1, is under the direction of Mr. Edwin G. Czarnecki. Mr. Czarnecki is the single executive responsible to JPL and to Boeing management for the accomplishment of the Voyager Spacecraft Phase IA, Task B work and will direct subsequent phases of the program. He reports directly to Mr. George H. Stoner, Vice President and Space Division General Manager.

Although Boeing has capability in all aspects of the Voyager Program it is planned to extend this capability in depth through association with companies recognized as specialists in technologies critical to Voyager

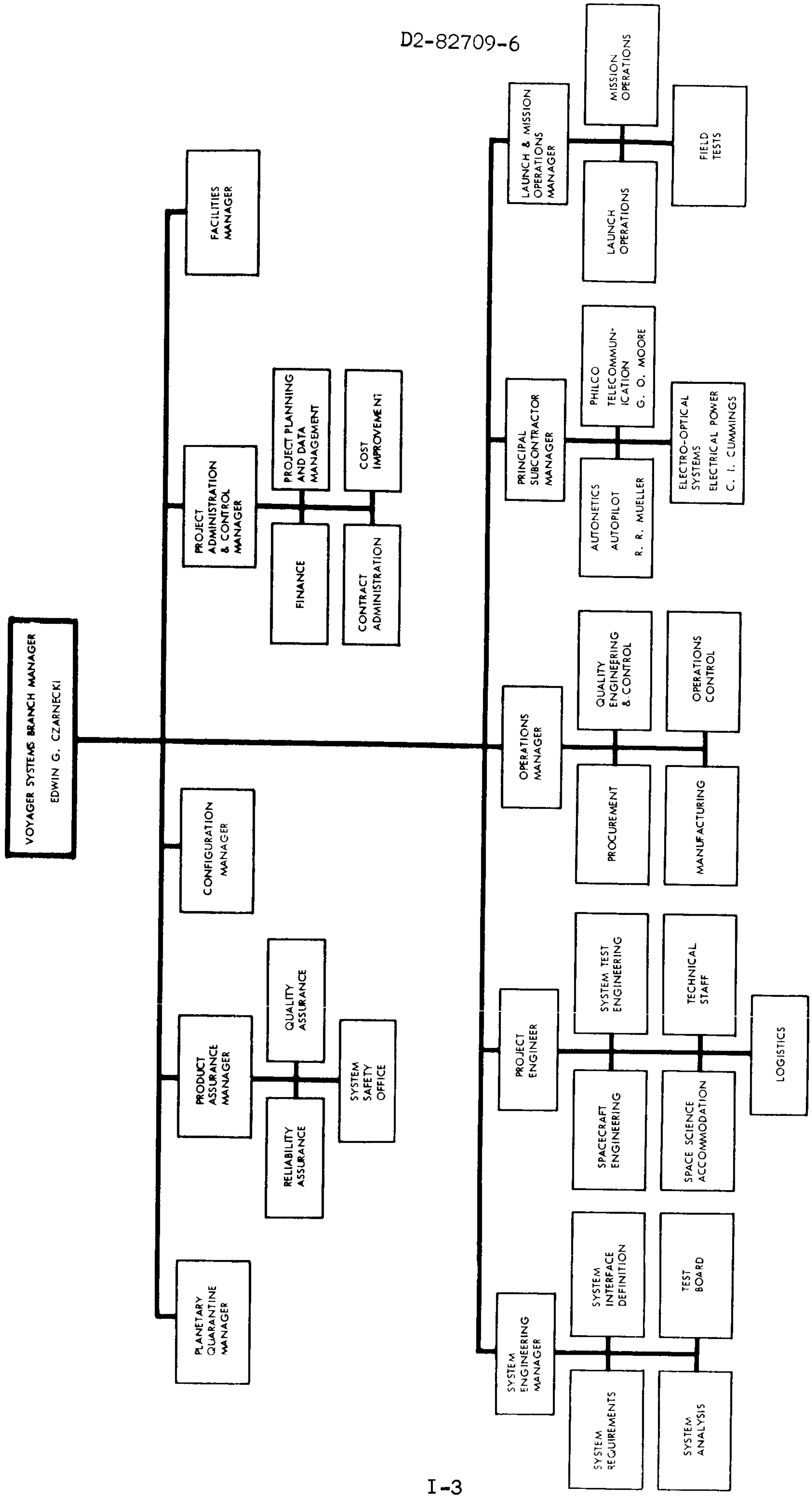


Figure I-1: Boeing Voyager Spacecraft System Organization performance. The following team members have been chosen because of their experience and past performance:

Autonetics, North American Aviation, Anaheim, California

Autopilot and Attitude Reference Subsystem
Mr. R. R. Mueller, Program Manager

Philco, Western Development Lab, Palo Alto, California

Telecommunication Subsystem
Mr. G. C. Moore, Program Manager

Electro-Optical Systems, Inc., Pasadena, California

Electrical Power Subsystem
Mr. C. I. Cummings, Program Manager

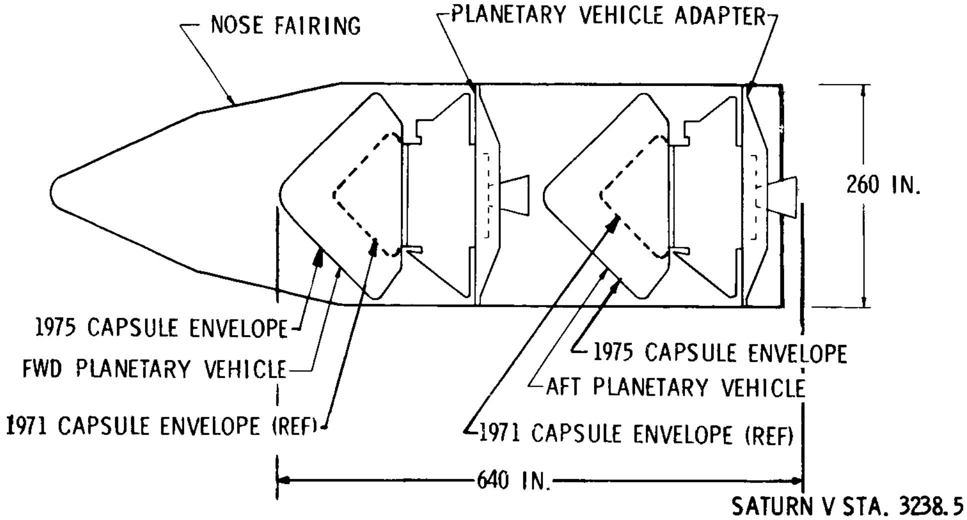
These subcontractor team members have been associated with Boeing on the Voyager Program for periods of 7 to 14 months. As a result of this, there has been sufficient exchange of information to make possible immediate implementation of the project with a Boeing team capable of satisfying the JPL requirements.

The preliminary design approach by the Boeing team has emphasized

- High probability of mission success.
- Conservatism, simplicity, selective redundancy in critical areas, and the use of Mariner experience.
- Versatility to accommodate a wide range of payload, mission, and data requirements.

The Voyager Flight Spacecraft, shown in Figure I-2, has the following principal features:

1) A capability to meet or exceed all mission requirements established in the Voyager 1971 Preliminary Mission Description.

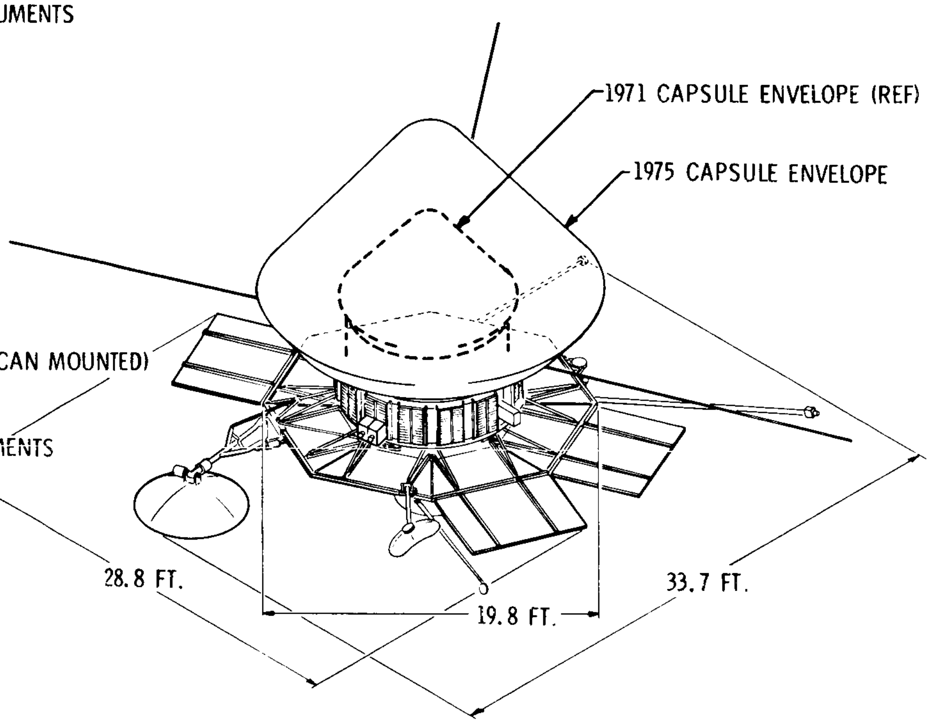


TOTAL SATURN V PAYLOAD ARRANGEMENT

CANDIDATE SPACECRAFT
SCIENCE PAYLOAD

SPACECRAFT BODY - MOUNTED INSTRUMENTS
- PLASMA PROBE
- COSMIC RAY TELESCOPE
- COSMIC DUST DETECTOR
- TRAPPED RADIATION DETECTOR
- ION CHAMBER
- MAGNETOMETER
- RF NOISE DETECTOR
- IONOSPHERE SOUNDER
- BISTATIC RADAR
- GAMMA RAY
- GRAVIMETER
- ULTRAVIOLET SPECTROMETER (SCAN MOUNTED)

SCAN PLATFORM - MOUNTED INSTRUMENTS
- INFRARED SPECTROMETER
- INFRARED SCANNER
- PHOTOIMAGING DEVICE



ASSEMBLED VIEW OF PLANETARY VEHICLE

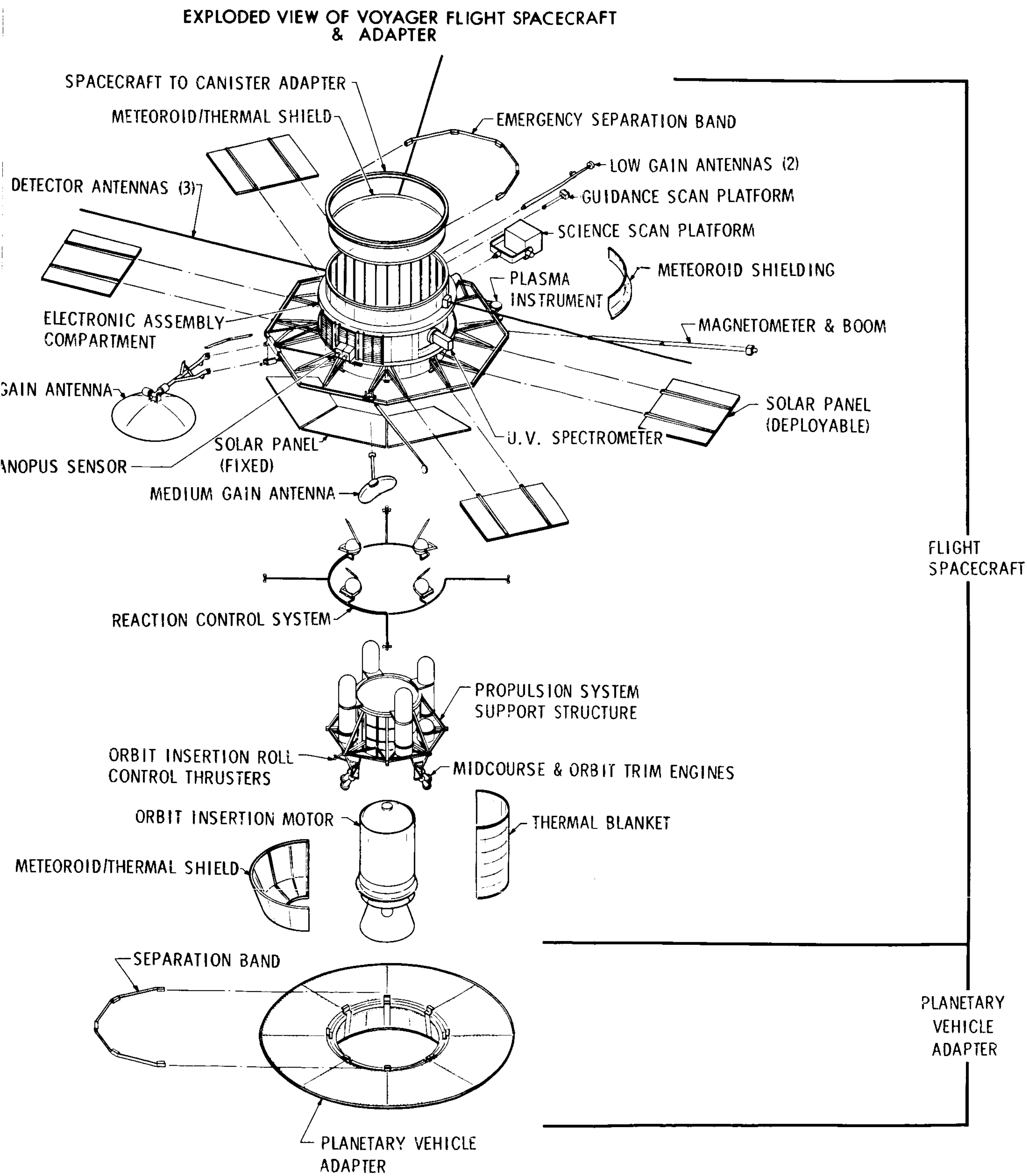


Figure I-2: Voyager Mars Mission Configuration

2) A high probability (approximately 80 percent) of returning science data from at least one spacecraft in Mars orbit. The reliability of the Spacecraft Bus improved from .82 in Task A to .90 in Task B, primarily because of additional redundancy in the telecommunications system.

3) A spacecraft with subsystems sized to accommodate the range of anticipated Mars missions. The 1971 mission capability includes a 93-day launch period, periapsis altitudes as low as 400 km, orbit periods as low as 2.8 hours, and solar occultations as low as 3.7 hours.

4) A single propulsion module capable of fulfilling all Mars mission propulsion requirements from 1971 through 1977 without resizing or changing the propellant quantity.

5) Electrical and electronic systems designed so that no single failure will cause a catastrophic effect on the mission.

6) A computer and sequencer designed so that completion of a nominal mission can be accomplished with programs stored on-board and without ground command intervention unless required by trajectory dispersions or biasing. The ground system can override and back up these programs and command midcourse and orbit corrections when necessary.

7) Space is provided for 16 standard equipment assembly packages, 16" x 32" x 8.5", fastened to the 10-foot-diameter cylindrical structure and thermally interconnected. Fourteen of these are used in the preferred design, all of which employ standardized internal packaging. Thermal control of these assemblies is by space-facing plates radiating through Mariner C type bi-metallic-actuated louvers.

The Flight Spacecraft is 28 feet 10 inches wide, solar panel tip to solar panel tip. The height is 158" compared to a maximum allowable of 208 inches. The estimated weight is 1920 pounds for the Spacecraft Bus. A contingency of 180 pounds is therefore available for selective use during the detail design phase. The estimated weight of the propulsion module is 14,840 pounds with a contingency of 160 pounds (approximately 10 percent of the inert weights) available for use during the design phase.

Analyses and tradeoffs of the four specified Flight Spacecraft propulsion systems indicated that they were nearly equivalent in meeting the JPL specified requirements. The propulsion system selected is the modified Minuteman Wing VI second stage motor for orbit insertion and a hydrazine subsystem using four 200-pound thrust engines for trajectory corrections, and for orbit trim and vernier. The choice of this selected system was based primarily on the lower technical risk in the development of this system and the larger weight available for reallocation. In addition, it makes maximum use of available proven hardware.

A trade study was conducted between propulsion systems sized for 1971, 1973, and 1975, 1977. The study showed that there were only minor differences and that a single design can be developed, tested, and used without change for all missions, 1971 through 1977.

Wide variations in mission requirements are accommodated by the combined use of the solid motor augmented by the hydrazine system for orbit vernier. The performance of the selected propulsion unit exceeds all 1971 mission specification requirements. It provides an orbit insertion velocity increment in 1971 of 2.39 km/sec (2.2 km/sec design goal) with the 2000-pound capsule.

The hydrazine engines selected for trajectory and orbit correction maneuvers utilize a Shell 405 spontaneous catalyst. The engines are of the same type as those selected during the Task A preliminary design. They provide a total velocity change capability of 637 meters per second for the 1971 mission. The hydrazine subsystem has an engine-out capability without malfunction detection and switching. This is accomplished by canting the engines and using jet vane thrust vector controls to maintain the thrust vectors through the vehicle center of gravity. This, together with the use of proven components, results in a high confidence in the predicted reliability of 0.9960 for the preferred propulsion module.

The telecommunications subsystem is sized to meet the mission design requirements. It can accommodate higher data rates, and allow additional modes if such needs develop. The system selected uses a 50-watt traveling wave tube amplifier and a 6-1/2 foot diameter paraboloidal high-gain antenna with two axes of rotation. Complete coverage of Earth is provided during Earth-to-Mars transit, orbit insertion, and orbiting flight. Space is available for growth to an 8 x 12 foot paraboloid. A maximum data rate of 7500 bps is provided with the $6\frac{1}{2}$ foot diameter antenna. The system has the potential for a data rate of 15,000 bps for a period of 20 days after encounter under worst case conditions. A 1260 bps backup mode is available during the first 100 days of Mars orbit. This is accomplished with a fixed Mariner C paraboloidal antenna oriented to provide coverage of Earth during that period.

Five telemetry modes have been provided with data rates of 7500 bps for orbital use, 1260 bps for backup and late mission use, 80 bps for launch and interplanetary cruise, 1.64 bps for emergency use with the low-gain antenna, and an acquisition mode without data transmittal.

Data storage capacity is 3.8×10^8 bits in seven tape recorders. Recording and playback rates can be controlled redundantly through the Data Automation Equipment, Earth Command and the Computing and Sequencing Subsystem.

The Command Subsystem provides for two hundred (27-bit) stored and direct commands with growth provided for by expansion of the output combiner. Two complete, parallel command detectors and decoders with selection logic permits either detector to operate with either decoder to provide high reliability. The probability of executing a false command is several orders of magnitude less than the JPL requirement of 10^{-8}.

The Computing and Sequencing Subsystem controls the sequencing of time-dependent events during the Voyager mission. All functions for a nominal mission can be sequenced from launch through the end of orbital operations without command from mission control unless required by trajectory dispersions or biasing. The selected subsystem is a special-purpose programmable digital computer with an overall reliability of 0.986. It has a capacity for storage of 1024 (27-bit) words and a capability to execute 140 difference commands. Seven-hundred words of storage are required to perform mission functions leaving a 30-percent reserve capacity in a standard size core memory assembly.

The Guidance and Control Subsystem is similar to that selected in Task A and draws heavily on Mariner and Ranger concepts. The Canopus and Sun sensors, the analog type autopilot, and the cold nitrogen reaction control system maintain cruise attitude within ±0.3 degree. A planet sensor, limb detector, and terminator detector have been added to the Task A system. Single-axis ball and air bearing gyros and free rotor gas bearing gyros were re-examined. The free-rotor, gas-bearing Minuteman G6B gyro, modified to a higher torquing capability, was selected because of (1) demonstrated performance in the Minuteman application, and (2) a minimum number of units required for operational redundancy.

Reaction Control is by expulsion of cold nitrogen gas through coupled 0.125 pound pitch and yaw thrusters and coupled .035 pound roll thrusters. Sixteen separate thrusters are provided in a redundant configuration. Four titanium tanks contain 44 pounds of nitrogen. Under nominal conditions the nitrogen will last four years.

The Electrical Power Subsystem has been revised from the Task A design to satisfy new mission and physical constraints. Fixed and deployable panels were evaluated extensively. The selected solar panel array consists of 8 fixed trapezoidal panels (178 square feet), and 4 deployable rectangular panels (138 square feet) for a total of 316 square feet. This configuration meets power requirements for all mission periods and orbit selections, and in addition will meet major mission objectives if one panel fails to deploy. The solar electrical system provides 908 watts of gross power for spacecraft, capsule, and battery charging loads at the end of six months of orbital operation. The configuration can be tested in the Boeing Space Chamber with panels deployed.

Three silver cadmium batteries rated at a total of 2720 watt hours provide power for off-Sun periods up to 3.7 hours. Battery size and circuit design allow the mission to be completed if any one battery section fails. Prime power is distributed at 2400 cycles per second, single phase, 50 volts. Three sets of regulators, inverters, and switching equipment are provided in a redundant configuration. This provides capability to operate all vehicle subsystems in event of a failure of any one power channel. Redundant 400 cycle per second inverters are provided for scan platform controls. Redundant precision oscillators are also provided.

The spacecraft structural arrangement is extensively revised from the Task A preferred configuration because of the larger and heavier propulsion module and increased capsule attachment diameter. Structural weight is 385 pounds and consists of (1) the primary structure assembly; (2) the external supports for appendages; (3) the capsule support and emergency separation assembly; and (4) an eight point Planetary Vehicle separation assembly. The primary structure is a 120-inch diameter magnesium shell, 85 inches long, of conventional semi-monocoque design. This shell provides direct support for attachment of 16 equipment modules (14 used) and for distribution of thermal loads between the assemblies. The Planetary Vehicle adapter is designed to support the spacecraft at eight points and provides uniform loading at the nose fairing interface.

The mechanisms employed for release, deployment, and latching of deployed booms or linkages are the same as those proposed during Task A. Dual bridge-wire, pyrotechnic pin-pullers are used to release the pins holding the various components in their boost positions. Vinson-type actuators were selected for the deployment function; and spring-actuated, taper-pins are used to lock the components in their deployed positions. Self-aligning, spherical bearings are used for all hinge joints to counter any binding effects caused by thermal distortion; and sleeve bearings within the spherical bearings provide a second path of rotation, thus increasing the reliability of the system.

Four-segment, V-block separation bands are used to release the Planetary Vehicle from its adapter and also to effect emergency release of the Flight Capsule. Four pyrotechnic separation devices in each band assure a release reliability of .99992. Eight helical compression springs impart a total separation velocity of 1 foot per second.

The selected pyrotechnic subsystem follows the basic concept of the Mariner series in using capacitors and solid state switches. The pyrotechnic subsystem provides for a set of 21 command signals and 59 electro-explosive devices.

The Temperature Control Subsystem maintains the Spacecraft Bus, propulsion module and science instruments within specified operating temperatures throughout all the mission phases. The design approach, parts, and materials are similar to those used on Mariner C. The equipment modules are coupled thermally and temperature control is accomplished by 52 square feet of bi-metal actuated louvers and high emittance radiator surfaces. The thermal dissipation capacity of the system is approximately 1200 watts, providing nearly 50 percent more capability than required to maintain gross thermal balance.

A science scan platform (GFE) is postulated to support the following science equipment: Infrared Spectrometer, Infrared Scanner, and two television cameras. This platform, with two-axis gimbal drive, provides the science instruments with clear views of Mars. An Ultraviolet Spectrometer is mounted on the spacecraft body with adequate scanning capability.

Substantial additional study and analysis has been made of ways to meet the planetary quarantine requirements and of the resulting Flight Spacecraft design constraints. New data made available or developed since the Task A report are:

1) The new Martian atmosphere which affects both probability of capture and heating rate of contaminated ejecta.
2) Micro-organism IR emissivity which has been determined by Boeing to be approximately 0.2 instead of the previously estimated value of 0.5 to 1.0.
3) Increased microbial kill due to low ultraviolet attenuation in the Martian atmosphere.
4) Reduction by a factor of 10^4 in the meteoroid environment at Mars and associated reduction in the amount of contaminated material spalled off the orbiting spacecraft.
5) Tests run by Boeing which demonstrate with a high confidence that hydrazine is self-sterilizing.
6) Firings of solid engines by Boeing which indicate that the micro-organisms do not survive the hot firing.

Based upon the above, the approaches taken in each hardware area for the selected design are:

1) Micrometeoroid Ejecta--No surface sterilization is provided for the spacecraft, but study and analysis should be continued. The higher ultraviolet kill and the lower micrometeoroid environment reduces the probability of contaminating the planet to 2.8×10^{-5}.
2) Reaction Control and Thrust Vector Control Ejecta, Midcourse and Orbit Trim Pressurant--Sterilize or decontaminate the nitrogen, Freon, and hardware internal surfaces. Study further to assess ultraviolet kill.
3) Midcourse and Orbit Trim Engine--No sterilization of the propellant or propellant hardware in this system is provided because of hydrazine's self-sterilizing characteristics. Tests in Phase IB are required to verify that micro-organisms are not ejected from down stream hardware in Mars orbit.
4) Orbit Insertion Engine--Based upon UV kill and hot firing indications, this engine is not sterilized. Further analysis and hot test firings in Phase IB are required to confirm data prior to initiation of engine procurement.

The OSE selected is a modest extension of Mariner concepts. Subsystem test sets are used as the basic building blocks for the System Test Complex. The System Test Complex employs a Scientific Data Systems general purpose digital computer in a Central Data and Control System for automatic control of the subsystem test sets and central data analysis and display. The total design emphasizes minimum new development to enhance mission success and cost effectiveness.

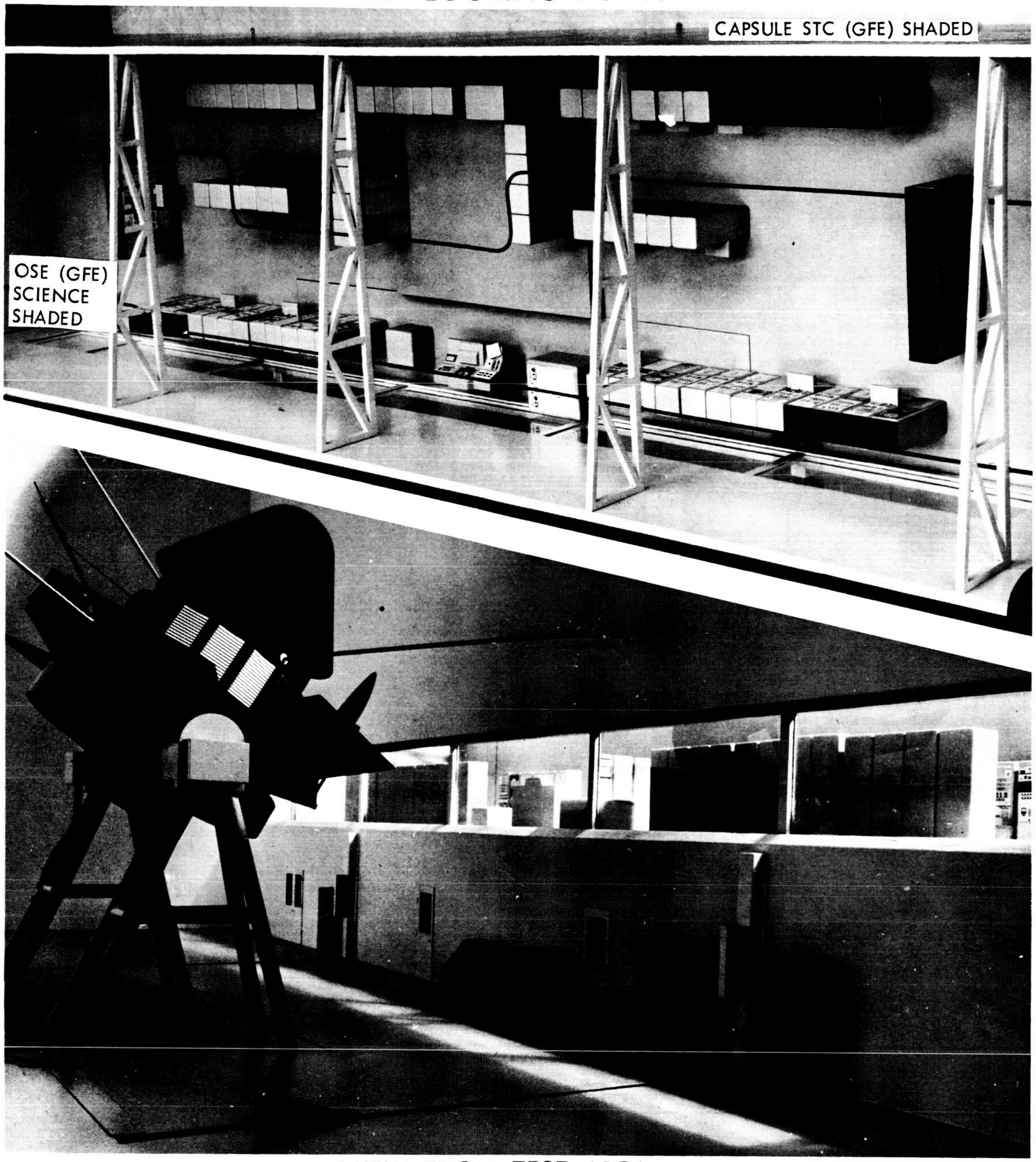
Several existing test systems were considered for use in System Test Complex design and traded off against the preferred concept which is an updated version of that proposed by Boeing in the Phase IA Task A submittal. Systems considered include the Apollo Acceptance Checkout Equipment (ACE) and the Mariner C test equipment. The trade studies indicate that use of ACE would be either non-responsive to specification requirements or, if subsystem OSE were incorporated into the system, would be unnecessarily complex. Mariner C equipment does not include the required degree of central control and automaticity.

All requirements can be met with the preferred design which is well within current technology. It is planned that existing hardware be employed to a maximum degree in defining the Spacecraft System OSE and common components be employed wherever feasible.

The building block approach to design has also been applied to computer program development. Mission operations and test programs are assembled from sub-routines prepared in standard format in accordance with standardized software requirements. This minimizes software development time and costs and allows computer program preparation in parallel with equipment design.

Subsystem Test Sets are typically 1 to 9 standard racks containing equipment similar to that used in the Mariner Subsystem OSE. When elements of these are integrated with the SDS 920 (or 930) computer and appropriate interface adapters, they form a System Test Complex (STC) of approximately 55 cabinets (racks, output data units, and control consoles). Addition of the Mars surface lander capsule and Science Subsystem OSE brings the total Planetary Vehicle System Complex (STC) to about 76 cabinets. Figure I-3 shows a model of the

VIEW LOOKING DOWN



VIEW FROM TEST AREA

Figure I-3: System Test Complex And Equipment

STC, typical test facilities, and equipment. Elements of the STC are employed as an integral part of Launch Complex Equipment (LCE).

A test team concept is planned wherein technical personnel experienced in spacecraft and OSE design, systems test operations, launch and mission operations, and spacecraft assembly and quality control will be formed into test groups. One of these teams will be assigned to each flight spacecraft and spare and will follow that vehicle from assembly through launch. Selected elements of the test team will continue to support mission operations for their spacecraft.

The Task B review and revision of the preliminary design for the Voyager Spacecraft System has emphasized conservative design, particularly in the use of proven equipment and techniques to the greatest extent consistent with system requirements. High reliability has been achieved through selection of space-proven components and through design of redundant capabilities into subsystems and equipment. The propulsion subsystem has been sized to achieve a range of flight trajectories and Mars orbits for missions in the years 1971 through 1977. The preferred Flight Spacecraft design provides mission versatility and capability for growth. As a result of the Task B activities, The Boeing Company has developed a design believed to be optimum for achieving objectives of the Voyager 1971 mission.

CONTENTS

		PAGE
1.0	VOYAGER 1971 MISSION OBJECTIVES AND DESIGN CRITERIA	1-1
1.1	Program Objectives	1-1
1.2	1971 Mission Objectives	1-2
1.3	Mission Constraints	1-3
1.4	Design Criteria	1-4
1.5	System Elements and Weight Allocation	1-4
1.6	Competing Characteristics	1-7
1.7	Mission Trade Study Summaries	1-9

1.0 VOYAGER 1971 MISSION OBJECTIVES AND DESIGN CRITERIA

This section discusses mission objectives and design criteria and their influence on the selected design. JPL's "Voyager 1971 Preliminary Mission Description" was used as the baseline for the mission objectives and design criteria summarized in this document. Although data were verified and expanded, no exceptions were taken to the "Voyager 1971 Preliminary Mission Description."

1.1 PROGRAM OBJECTIVES

Voyager Program Objectives--The primary objective of the Voyager program is scientific investigation of the solar system by means of instrumented, unmanned spacecraft that will fly by, orbit, and land on the planets. Scientific information about the origin, evolution, and nature of life will be acquired and applied toward an understanding of terrestrial life.

Voyager Mars Objectives--The primary objective of the Voyager missions to Mars beginning in 1971 is to obtain, through unmanned experiments on the surface of and in orbit about the planet, information of the planet's terrestrial life; atmospheric, surface, and body characteristics; and environment. A secondary objective is to increase the knowledge of the interplanetary medium between the orbits of Earth and Mars around the Sun by obtaining scientific and engineering measurements while the Planetary Vehicle is in transit.

Meeting both objectives necessitates an orderly program of continually improving knowledge in science and technology; therefore, each mission will be postured with objectives to expand that knowledge and a spacecraft system to achieve those objectives, all sequenced to take best advantage of preceding mission results.

1.2 1971 MISSION OBJECTIVES

The 1971 mission is first in the logical development of the sequenced missions for achieving Voyager Mars objectives. The 1971 mission objectives, postured to expand knowledge gained in the Ranger and Mariner programs, are:

1) Acquisition of scientific data from measurements of solar system phenomena, particularly those of the nature of the Mars planetary surface, planetary atmosphere, body characteristics, and extraterrestrial life;
2) Verification of technological capability of placing scientific payloads in orbit about the planet Mars, supporting them for extended periods of operation, and communicating reliably the data obtained from scientific and engineering measurements.

Although the scientific experiments for 1971 are not defined, science observations have been postulated to provide spacecraft accommodation for potential experiments. Therefore, two categories of science instruments are assumed to be best suited for the expected science experiments:

1) Spacecraft Body-Mounted Instruments--plasma probe, cosmic-ray telescope, cosmic-dust detector, trapped-radiation detector, ion chamber, magnetometer, RF noise detector, ionosphere counter, bistatic radar, gamma-ray detector, gravimeter, and ultraviolet spectrometer (scan mounted);
2) Scan Platform-Mounted Instruments--infrared spectrometer, infrared scanner, and photoimaging devices (two television cameras).

The primary technological objectives of the 1971 mission are to place a science payload in orbit about Mars; conduct observations of Martian phenomena with this payload over extended periods of time; and transmit the data to Earth. Additional technological objectives are to enter the Martian atmosphere and land on the Martian surface, and conduct observations relating to critical Mars landing design parameters in orbit and in the atmosphere as required.

1.3 MISSION CONSTRAINTS

Mission constraints imposed on the Spacecraft System by the 1971 mission are schedule, planetary quarantine requirements, number of launches, and existing mission support facilities. These constraints are considered firm and not subject to trade studies.

Mission Schedule Constraints--The Mars opportunity places an absolute constraint on the mission schedule; consequently, all design, development, fabrication, testing, and deliveries will conform to the established mission schedule milestones. The launch period for the Voyager 1971 mission will not be less than 45 days. The minimum daily firing window will not be less than 2 hours.

Mission Planetary Quarantine Constraints--The probability that Mars is contaminated prior to calendar year 2021 as a result of any single launch will be less than 10^{-4}. Consideration will be given to the implications of this restraint on the S-IVB stage, the spacecraft, the capsule, and all emissions and ejecta.

Number of Launches Constraints--One space vehicle will be launched during the 1971 Mars opportunity.

Mission-Support Facilities Constraints--The Spacecraft System will be compatible with the launch facilities and Deep Space Network. Complex 39 at Kennedy Space Center will be used for launching the space vehicle during the 1971 Mars opportunity. Launch azimuths will not be less than 60 degrees east of north nor greater than 115 degrees east of north.

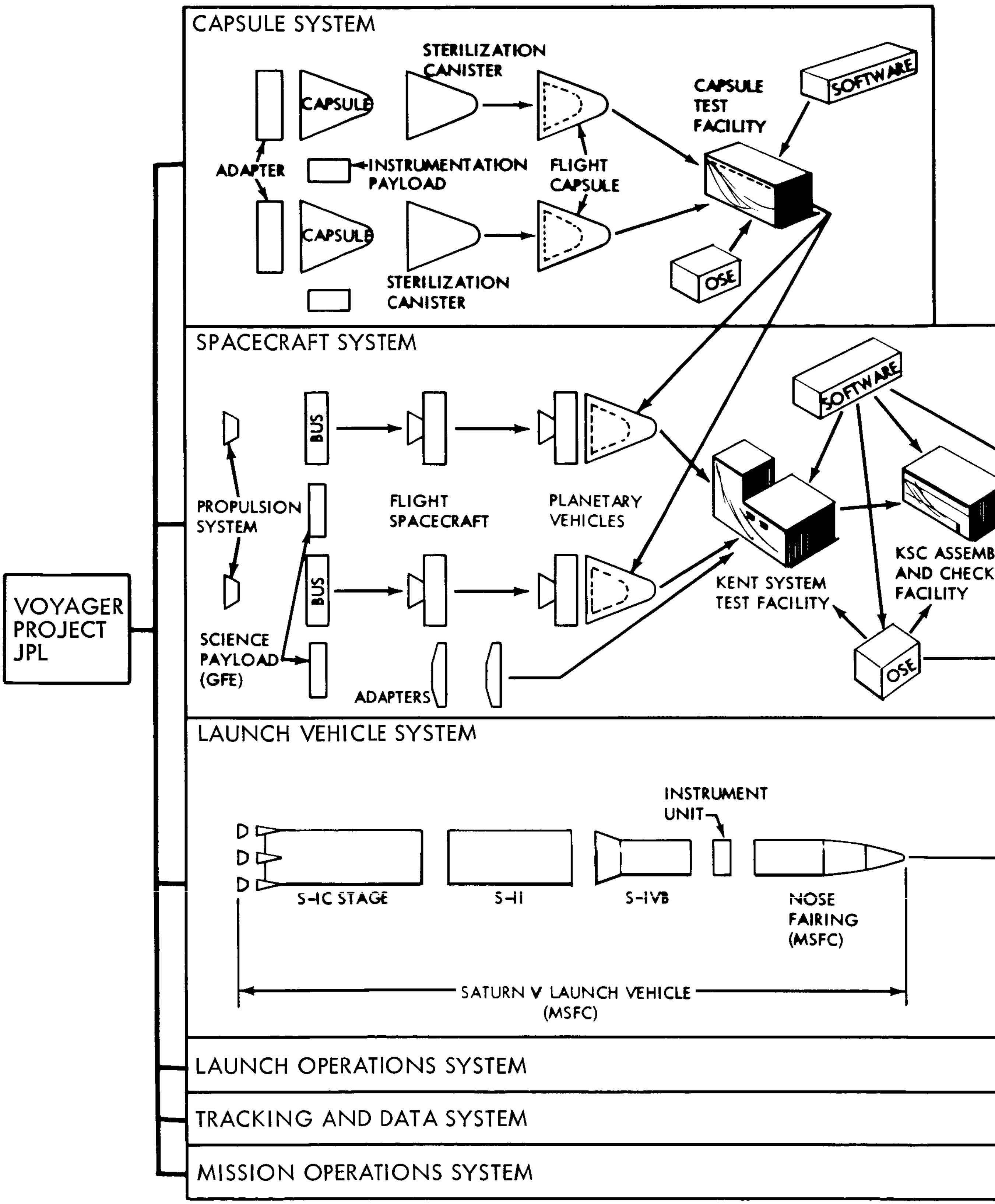
1.4 DESIGN CRITERIA

To develop a Spacecraft System with the highest probability of mission success, Boeing's design approach is to utilize space-proven concepts such as those developed in the Pioneer, Ranger, and Mariner programs. If space-proven or space-approved hardware exists and fulfills the functional requirements of the Spacecraft System concept, that hardware will be given precedence in design consideration. If new hardware must be developed, proven design concepts will be used that emphasize simplicity to ensure reliability and mission success.

Performance reliability should be included as an inherent feature of the system concept through conservative design, hardware environment control, strict adherence to procedural development, rigorous qualification testing at all levels, redundancy, and emphasis on human engineering.

1.5 SYSTEM ELEMENTS AND WEIGHT ALLOCATION

The principal elements for the 1971 mission are shown in Figure 1-1. Each system includes operational hardware, spares, software, and support equipment; management, personnel, and facilities for development and test of hardware and software; and support personnel and equipment for the prelaunch, launch, and flight phases of the mission.



VOYAGER PROJECT JPL
CAPSULE SYSTEM
STERILIZATION CANISTER
CAPSULE
ADAPTER
INSTRUMENTATION PAYLOAD
FLIGHT CAPSULE
CAPSULE TEST FACILITY
SOFTWARE
OSE
STERILIZATION CANISTER
SPACECRAFT SYSTEM
BUS
PROPULSION SYSTEM
FLIGHT SPACECRAFT
PLANETARY VEHICLES
KENT SYSTEM TEST FACILITY
KSC ASSEMB
AND CHECK
FACILITY
SCIENCE PAYLOAD (GFE)
ADAPTERS
LAUNCH VEHICLE SYSTEM
INSTRUMENT UNIT
S-IC STAGE
S-II
S-IVB
NOSE FAIRING (MSFC)
SATURN V LAUNCH VEHICLE (MSFC)
LAUNCH OPERATIONS SYSTEM
TRACKING AND DATA SYSTEM
MISSION OPERATIONS SYSTEM

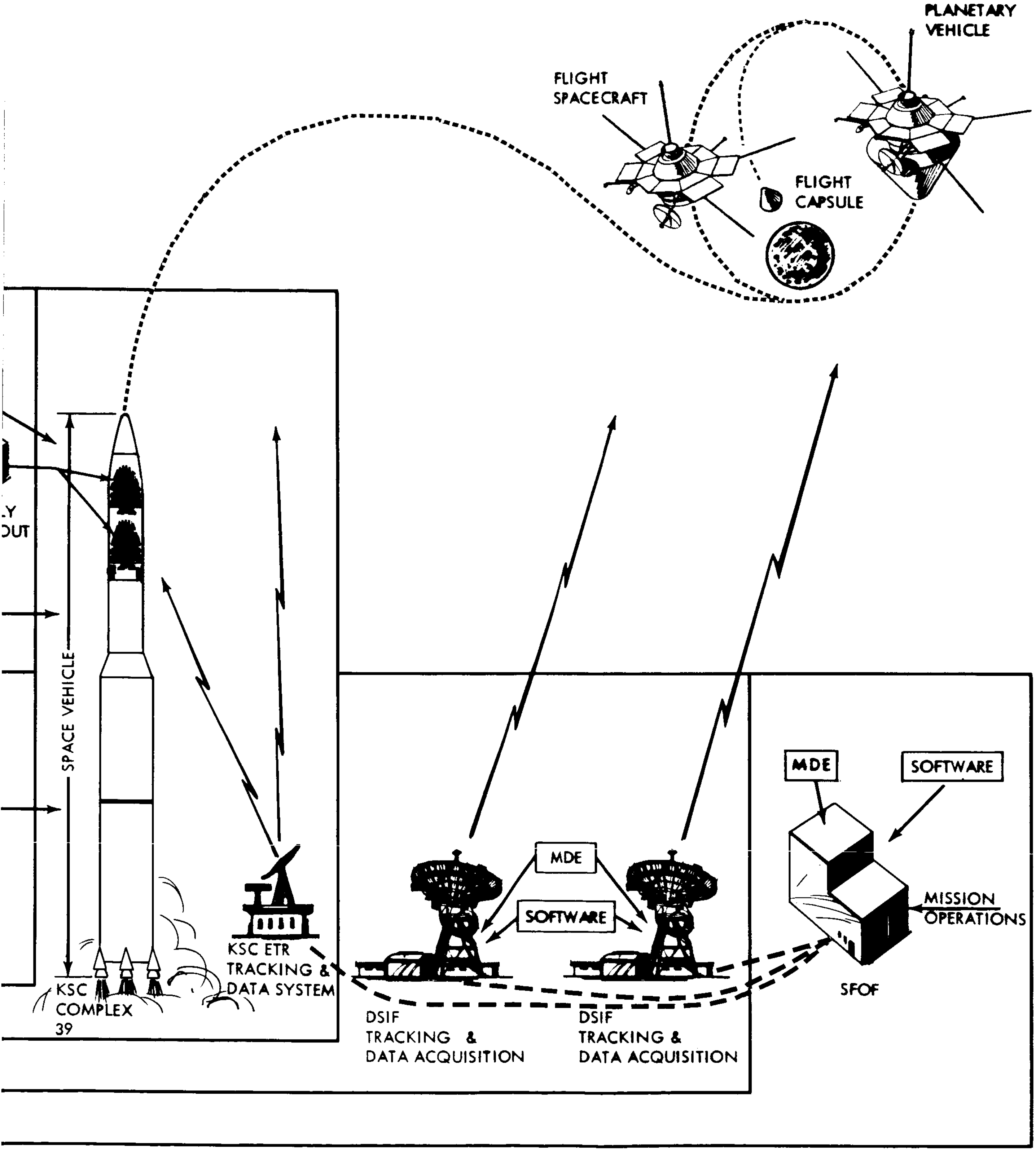


Figure 1-1: Mission System Elements

The gross payload capability of the Saturn V including the portion of the nose fairing injected into trans-Mars trajectory is 63,000 pounds for a launch azimuth of 115 degrees. Ample capability exists to launch the 1975 or 1977 mission with Flight Capsules weighing 10,000 pounds. Weights of the various elements of the 1971 mission are allocated as follows:

Space Vehicle (minus launch vehicle)					44,000 Lbs.
Planetary Vehicle Adapter				1,500 ea	
Planetary Vehicle				20,500 ea	
Flight Capsule *			3,000 ea ***		
Flight Spacecraft			17,500 ea		
Propulsion Module		15,000 ea			
Spacecraft Bus **		2,500 ea			

* A maximum of 250 pounds can remain with the Flight Spacecraft.

** Includes science payload weight of 400 pounds.

*** Per JPL, 2000 pounds are used for spacecraft design.

1.6 COMPETING CHARACTERISTICS

Boeing recognizes that the selection of a preferred Spacecraft System, which will provide the best assurance of successful operation, is maximized when all system trade studies are evaluated against a standard of competing characteristics. The standard provided by JPL is shown horizontally at the top of Table 1-1 with primary emphasis on "Probability of Success." Mission characteristics are arranged vertically with emphasis on "Overall System" performance. Each block of the matrix identifies the next level of competing characteristics for evaluating specific system trades as discussed in Section 3.11.

Table 1-1: Competing Characteristics and Evaluation Criteria

System Study \ Competing Characteristic	Probability of Success (Priority 1)	Performance of Mission Objectives (Priority 2)	Cost Savings (Priority 3)	Contributions to Subsequent Missions (Priority 4)	Additional 1971 Mission Capability (Priority 5)
Overall System	• Probability of Operational Success • Tech Risk • Schedule Risk • Reallocated Wt	• Min Impulse Bit • Design Goal Performance • Orbit Versatility	• System Cost Differences	• Growth	• Additional or Redundant Experiments • Additional Mission Capability
Mission Profile (Trajectory Selection)	• Orbit Lifetime • Occultation of Sun • Occultation of Canopus • Occultation of Earth	• Occultation of Sun • Occultation of Earth • Observation of Darkening Phenomena • TV Coverage • Mars Orbit	• Trip Time	• Observation of Future Landing Sites • Atmospheric Data	• Flexibility
Propulsion Subsystem	• Previous Space Experience • Compatibility with Mission Environment • Reliability • Safety	• Velocity Increment for Orbit Insertion	• Total Propulsion Cost • Ease of Sterilization	• ΔV Capability vs Future Requirements	• ΔV Above 1971 Requirements
Other Subsystems	• Reliability	• Performance Parameters	• Total Subsystem Cost	• Performance Growth vs Future Requirements	• Performance Above 1971 Requirements
Spacecraft Bus	• Reliability • Technical Risk • Complexity • Mariner Exper.	• Performance Accuracy	• Total Subsystem Cost	• Volume, Weight, Power, & Structural Margins	• Weight, Power, & Volume Margins for Additional Missions
Operational Support Equipment	• Development Risk • Practicality • Probability of Checkout • Data Correlation • Hardware Availability		• Skills Requirement • Training Requirement • Total Subsystem Cost	• Growth	• Flexibility

1.7 MISSION TRADE STUDY SUMMARIES

The primary objectives of the Task B mission trade studies were to develop data for program direction and to determine that all important factors pertinent to the development of mission objectives and design criteria were identified, evaluated, collated, and correlated. These objectives and criteria were established during Task B with the realization that Jet Propulsion Laboratory will ultimately determine or approve basic mission decisions. However, a valid comparison of alternate Spacecraft System designs required that a common, though preliminary, mission basis be established.

Trade study subjects have been established by examining the JPL-defined mission operational phases, derived functional flow diagrams, and major system elements for problem areas that require trade study evaluation. Each trade study has been made by taking the known, significant alternate solutions and evaluating them in terms of the competing characteristics established by JPL. The selected solution best satisfies the priority order of the competing characteristics. The mission trade studies conducted during Task B, including the candidate solutions, their evaluation, and the selected approach, are shown in Table 1-2. A further discussion of the factors influencing the selections is found in Sections 3.1 and 3.2. Table 1-3 is an example of a trade study summary sheet showing data for one of the mission trades. Additional trade studies will be conducted in later program phases as the mission definition and functional analyses are further refined.

B L A N K

Table

Trade Study Title		Trade Study Alternate Soluti	
		1	2
1.1	Mars orbit for second Planetary Vehicle after successful first vehicle	Orbit identical to that of first planetary vehicle	Similar orbit with minor variations
1.2	Planetary Vehicle, Mars differential arrival	Bias maneuver early in mission	Bias maneuver late in mission
1.3	Planetary Vehicle, Mars arrival date biasing	One-vehicle biasing maneuver	Two-vehicle biasing maneuver
1.4	Modes and sequences for separation of Planetary Vehicles	Separate in rapid sequence, activate simultaneously	Separate and activate forward vehicle, then second vehicle
1.5	Flight Capsule deorbit location	Deorbit at periapsis	Deorbit at apoapsis
1.6	Flight Capsule canister separation mode	Separate at least 10 days prior to Mars orbit injection	Separate 1 day prior to injection into Mars orbit
1.7	Periapsis trim	Thrust at apoapsis	Nonapoapsis

/

-2 Mission Trade Study Summary

ons	Selected Approach	S
3		
Major variations to alter surface coverage	2	Injection into similar orbits p data accuracy through redundanc ΔV, and less trajectory error; of mission objectives.
	1	When biasing maneuver is done e accomplished with smaller ΔV an can be integrated with midcours antenna.
	1	It is a Solution requirement that Procedure 1 was selected on a p success of only one of the two arrives in Mars orbit with smal orbit maneuver usage. Together flexibility and performance cov
Separate forward vehicle and <u>start</u> activation before separating aft vehicle	3	This sequence allows sufficient for malfunction decision. Yet vehicle battery capability.
Deorbit within 90° after apoapsis passage	3	Deorbit at this point permits r nominal terminal descent rate i which are favorable both to dat achieved without severe ΔV req
Separate after injection into orbit around Mars	1	Separation of the canister in t Early separation exposes capsul this probability is slight. Ea craft position fix before encou event is unsuccessful.
	1	Very small altitude change requ adjustment preferable, for larg apoapsis leads to smaller net e

v

election Rationale

rovides the greatest probability of experiment success and
y and assessment of first vehicle results, lower injection
thereby maximizing the probability of successful completion

arly, orbit determination and midcourse correction can be
d a higher probability of mission success. Biasing maneuvers
e maneuvers and monitoring data may be obtainable on low-gain

both vehicles have the capability of bias maneuvering.
erformance basis and affects the probability of mission
Planetary Vehicles. The vehicle with the unbiased trajectory
ler residual errors to correct and greater propellant for
the two vehicles have the potential of greater mission
erage.
separation distance to avoid collision and monitoring time
ts sequential time requirements are compatible with the

eduction in deorbit trajectory maneuver sensitivities and a
n sunlight, needed for the visual sensors. Trajectories
a acquisition and communication with the orbiter are readily
uirements.

he Mars orbit increases the probability of Mars contamination.
e to indirect contamination and meteoroid damage; however,
rly separation allows more time to obtain an accurate space-
ater; also it will permit corrective action if the separation

rements make the ΔV resolution errors by off-apoapsis
er periapsis altitude changes, the lower ΔV required at
rrors and more propellant left for subsequent maneuvers.

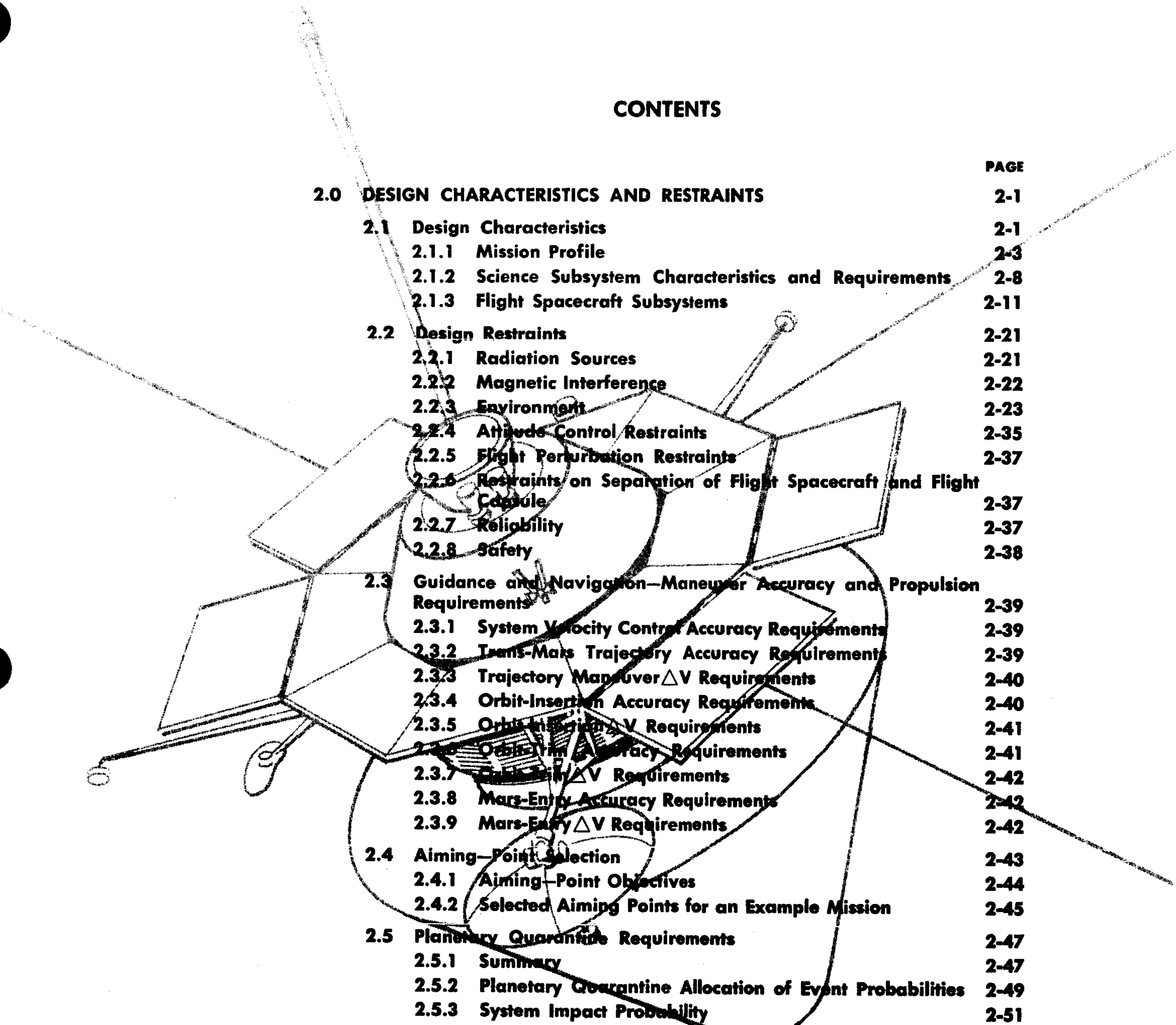
TABLE 1-3: MISSION TRAD

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE MAR
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS & COMPETING CHARACTERISTICS		ALTERNATE 1 ORBIT IDENTICAL TO THAT OF FIRST PLANETARY VEHICLE.
The second Planetary Vehicle must be able to achieve essentially the same orbit as planned for the first Planetary Vehicle, although it must arrive no less than 10 days later than first vehicle. FIRST PLANETARY VEHICLE ORBIT REQUIREMENTS 50-year orbit lifetime 30^{o} to 60^{o} orbit inclination. Periapsis location: o Southern hemisphere o 10^{o} -30^{o} from terminator in lighted hemisphere. No occultation of Earth, Sun, or Canopus prior to insertion and only minimum occultation of Sun and Canopus during orbit. Adequate time for communication with Earth Motion (posigrade or retrograde) such that capsule will make descent in sunlight when impact is near subperiapsis point. 2.2 kilometers per second ΔV available on Planetary Vehicle for insertion into orbit around Mars.	To what extent should the orbit of the second Planetary Vehicle differ from that of the highly successful first Planetary Vehicle? Second Planetary Vehicle enhances probability of mission success through redundancy. After a successful first vehicle, second vehicle may be inserted into a nearly identical orbit for acquisition of multiple data for statistical analysis, resolution of additional parameters, etc, or into a differing orbit for broader coverage. Selection parameters (derived from the competing characteristics); are: o Reliability o Performance	Establish second Planetary Vehicle in orbit identical to that of firs vehicle except for different epoch DISCUSSION Pro o High probability of individual experiment success and data accuracy through redund ancy. o Resolution of short-period variations in observed phenomena. Con o May impose awkward constrain on orbit injection in light variation in approach geomet and velocity. o No broadening of capability over primary objectives.

E STUDY SUMMARY (EXAMPLE)

S ORBIT FOR SECOND PLANETARY VEHICLE UPON SUCCESS OF FIRST PLANETARY VEHICLE			
MATRIX OF DESIGN APPROACH			SELECTION
	ALTERNATE 2 SIMILAR ORBIT - MINOR VARIATIONS IN ORBIT ELEMENTS	ALTERNATE 3 MAJOR VARIATIONS TO ALTER SURFACE COVERAGE	
e st n. d- ts of ry	Inject into orbit that achieves essentially the same surface coverage but differs from first orbit due to: variations in approach geometry and velocity; and minor variation in some orbital elements as indicated by success or failures of individual experiments aboard first vehicle. DISCUSSION Pro o Maximum probability of individual experiment success and data accuracy through redundancy and assessment of preliminary first flight results. o Lower injection ΔV with inherently smaller trajectory errors. Con o Only limited broadening of coverage.	Change orbit sufficiently to change initial coverage to other hemisphere and other limb (major change in periapsis location) and major change in inclination, nodal point, direction of orbital motion (retrograde, posigrade), etc. DISCUSSION Pro o Broaden capabilities o Contribute to subsequent missions Con o Limited reliability of individual experiments through redundancy. o Usually larger injection ΔV requirements with inherently larger trajectory errors.	Reliability 2, 1, 3 Performance 2, 1, 3 SELECTED APPROACH 2

CONTENTS



		PAGE
2.0	DESIGN CHARACTERISTICS AND RESTRAINTS	2-1
2.1	Design Characteristics	2-1
2.1.1	Mission Profile	2-3
2.1.2	Science Subsystem Characteristics and Requirements	2-8
2.1.3	Flight Spacecraft Subsystems	2-11
2.2	Design Restraints	2-21
2.2.1	Radiation Sources	2-21
2.2.2	Magnetic Interference	2-22
2.2.3	Environment	2-23
2.2.4	Attitude Control Restraints	2-35
2.2.5	Flight Perturbation Restraints	2-37
2.2.6	Restraints on Separation of Flight Spacecraft and Flight Capsule	2-37
2.2.7	Reliability	2-37
2.2.8	Safety	2-38
2.3	Guidance and Navigation—Maneuver Accuracy and Propulsion Requirements	2-39
2.3.1	System Velocity Control Accuracy Requirements	2-39
2.3.2	Trans-Mars Trajectory Accuracy Requirements	2-39
2.3.3	Trajectory Maneuver △V Requirements	2-40
2.3.4	Orbit-Insertion Accuracy Requirements	2-40
2.3.5	Orbit-Insertion △V Requirements	2-41
2.3.6	Orbit-Trim Accuracy Requirements	2-41
2.3.7	Orbit-Trim △V Requirements	2-42
2.3.8	Mars-Entry Accuracy Requirements	2-42
2.3.9	Mars-Entry △V Requirements	2-42
2.4	Aiming—Point Selection	2-43
2.4.1	Aiming—Point Objectives	2-44
2.4.2	Selected Aiming Points for an Example Mission	2-45
2.5	Planetary Quarantine Requirements	2-47
2.5.1	Summary	2-47
2.5.2	Planetary Quarantine Allocation of Event Probabilities	2-49
2.5.3	System Impact Probability	2-51
2.5.4	Meteroid Impact Ejecta Considerations	2-52
2.5.5	Propulsion and Reaction Control Subsystems	2-61
2.5.6	Conclusions	2-67

2.0 DESIGN CHARACTERISTICS AND RESTRAINTS

This section describes the design characteristics of the Flight Spacecraft as well as the applicable system requirements and restraints. The guidance and navigation requirements and restraints imposed on aiming-point selection are also discussed along with the planetary quarantine requirements.

2.1 DESIGN CHARACTERISTICS

The Voyager design is based on an extension of the Mariner IV design modified to permit insertion of each Planetary Vehicle into Mars orbit, subsequent orbital operations of each Flight Spacecraft, and entry, descent, and landing of each Flight Capsule.

The Spacecraft System includes the Flight Spacecraft, adapters, Spacecraft System test complex, special test facilities required for spacecraft and Planetary Vehicle testing, facilities at Kennedy Space Center used to assemble and prepare the Flight Spacecraft and Planetary Vehicle for launch; Flight Spacecraft OSE (including launch checkout equipment); developmental models; the Flight Spacecraft software and spares; and personnel required to support the Flight Spacecraft functions.

The Flight Spacecraft will be an attitude stabilized vehicle that uses the Sun and the star Canopus as celestial references. It will derive power from photovoltaic cells during solar-oriented periods and from batteries during boost and other periods during which the spacecraft is not oriented toward the Sun. The Flight Spacecraft will include a two-way communication system that will transmit data to and receive commands from Earth.

The Flight Spacecraft will carry a science payload capable of accomplishing the science mission postulated in Section 2.1.2, and will either transport a Flight Capsule to Mars or will perform the designated mission if it is launched without a Flight Capsule. The spacecraft will supply power, telemetry, data handling, and command transmission to the Flight Capsule. Upon ground command, the Flight Spacecraft will orient the Planetary Vehicle in any direction and initiate the separation sequence.

The Flight Spacecraft will include a modular propulsion unit that will perform trajectory corrections, orbit insertion, and orbit trim maneuvers.

Interplanetary Trajectory Corrections--To meet the 10-day arrival date separation requirement between the two Planetary Vehicles, and to perform all necessary interplanetary trajectory corrections and trajectory biasing, a total velocity increment capability of 200 meters per second will be provided for each Planetary Vehicle.

Mars Orbit Insertion--The minimum velocity increment for Planetary Vehicle Mars-orbit insertion will not be less than 2.0 kilometers per second, with a design goal of 2.2 kilometers per second.

Mars Orbit Trim--A total velocity increment of 100 meters per second will be provided to the total mass in Mars orbit prior to Flight Capsule separation.

The trajectory for the 1971 mission will be established based on the following constraints:

1) Type I transfer trajectories that have a maximum C_3 of 25 km^2/sec^2 will be used.

2) Hyperbolic excess velocity at Mars will not exceed 4.5 kilometers per second;

3) The absolute value of DLA will not be less than 5 degrees, and the inclination of the heliocentric transfer plane to the ecliptic plane will not be less than 0.1 degree;

4) Flight Spacecraft orbit insertion and the Flight Capsule deorbit maneuver will occur within view of the Deep Space Instrumentation Facility (DSIF) at Goldstone, California;

5) The separation between Planetary Vehicle arrival dates at Mars will not be less than 10 days.

2.1.1 Mission Profile

The Voyager 1971 nominal mission consists of four major phases: launch and transit, Planetary Vehicle orbit achievement, Mars entry, and Flight Spacecraft orbital operations. Each phase with its respective events, is illustrated in Figure 2-1 (see Section 3.9 for functional sequence details).

The following phase events are grouped into distinctive modes of operations together with allocated probability of success goals.

Launch and Transit Phase--Launch to an Earth-orbit altitude of 100 nautical miles is performed using the Saturn V launch system. Following S-II burnout, the forward nose fairing is jettisoned at approximately 350,000 feet. After Earth-orbit coast, a 2 to 90 minute variable, the S-IVB stage is restarted and both Planetary Vehicles are injected into a heliocentric trajectory. The probability of a successful launch and injection is 0.85, as a goal.

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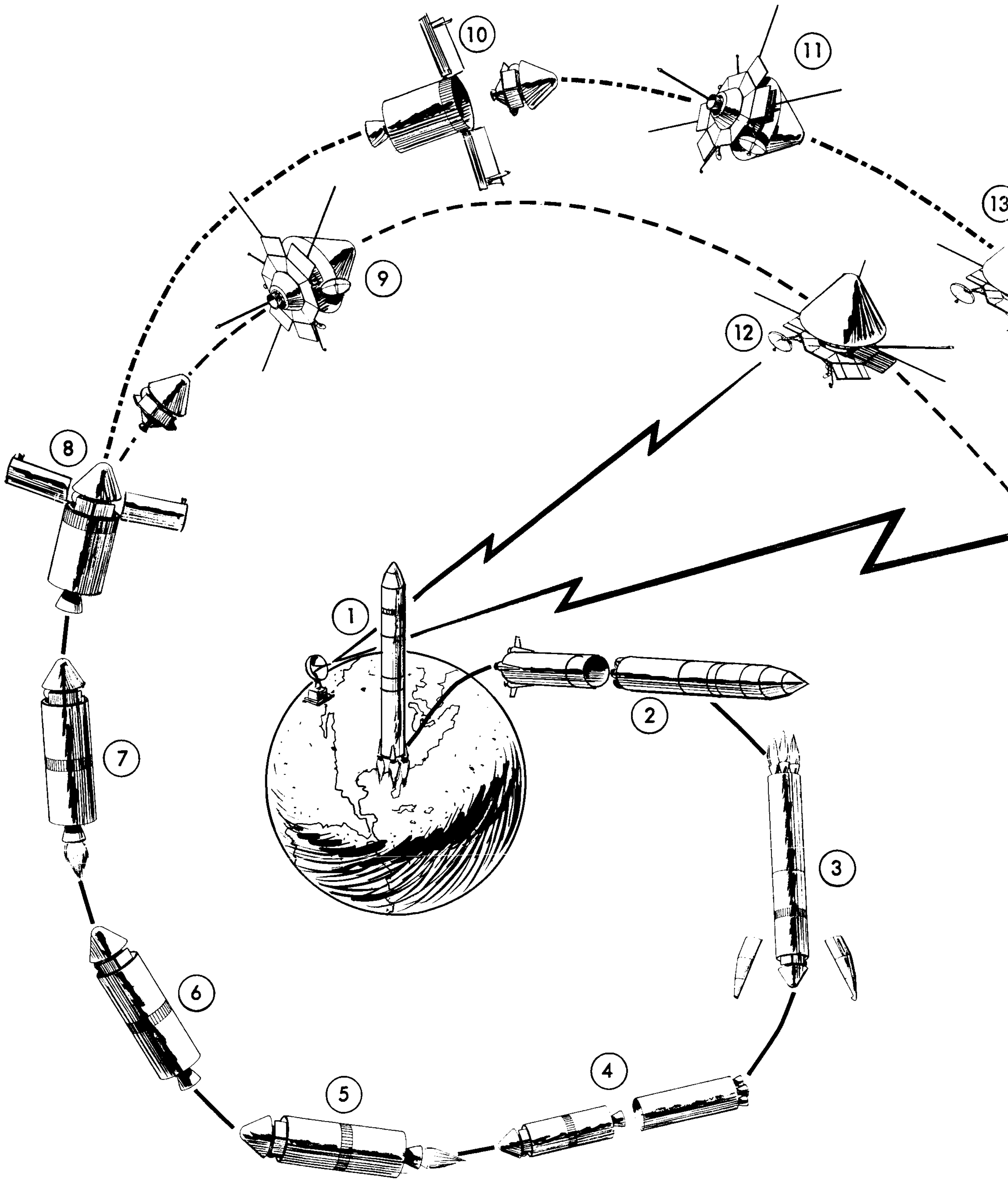

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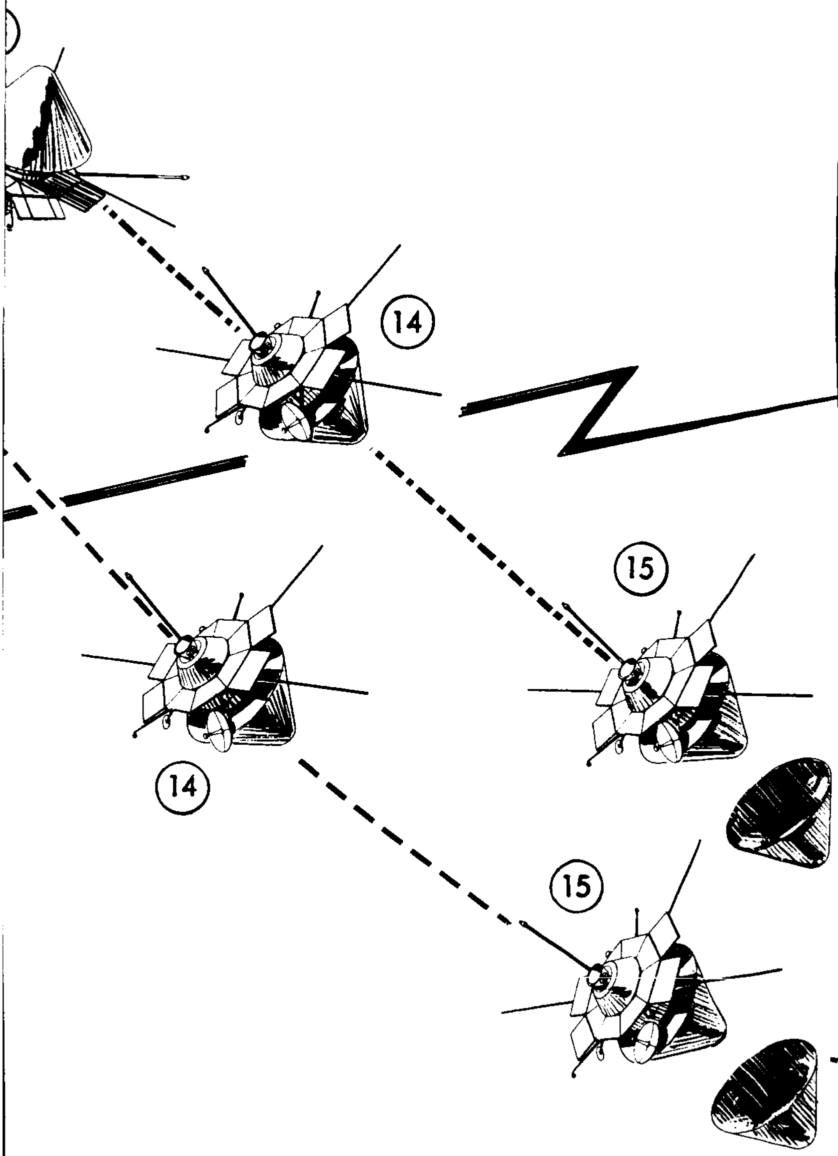

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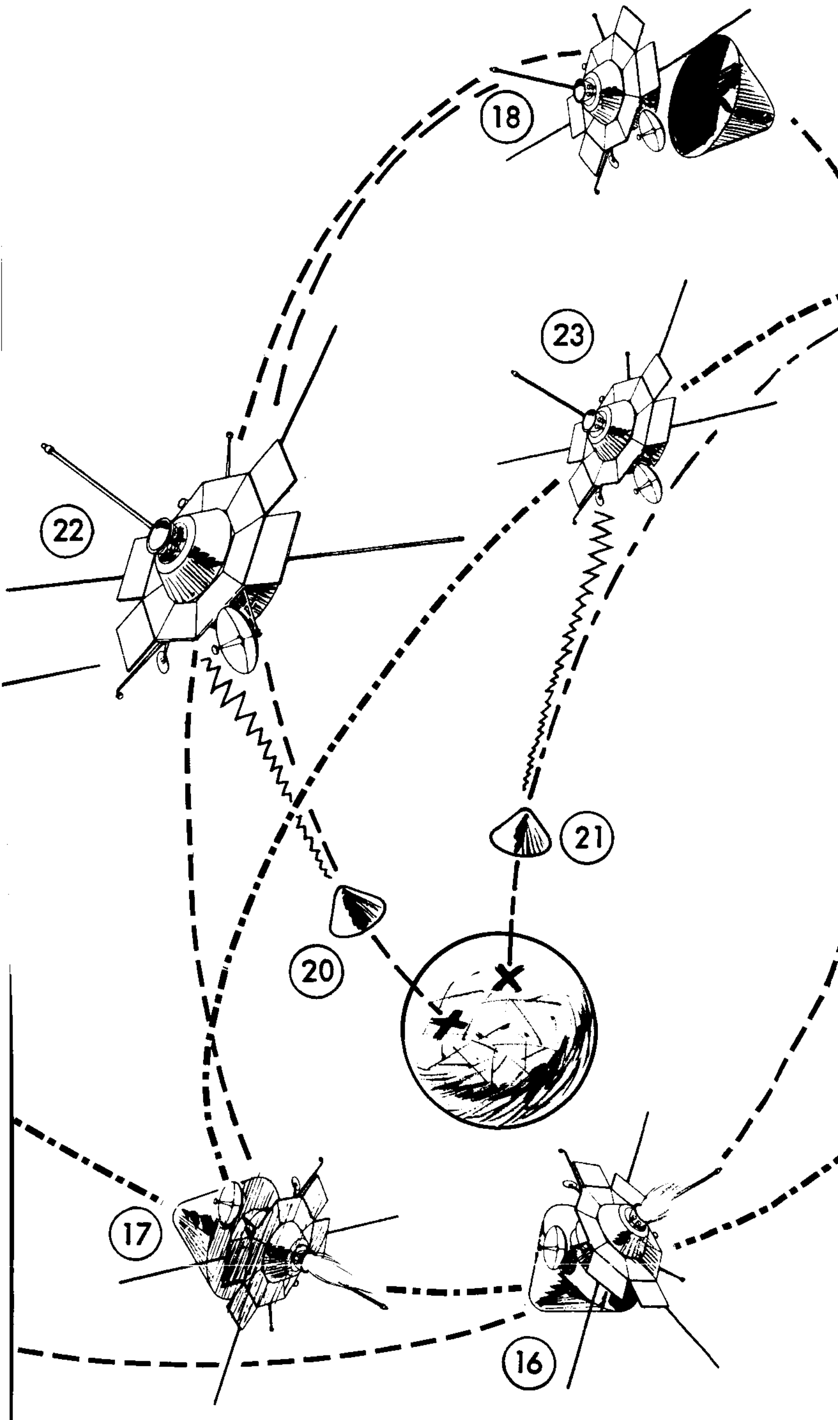
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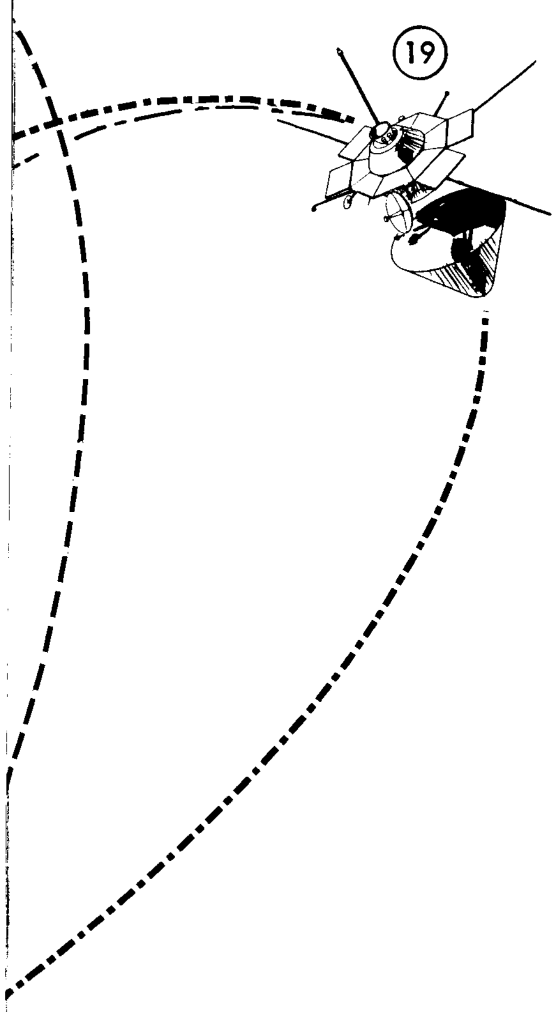


LAUNCH AND TR

1 SPACE
2 S-IC
3 S-II E
FAIRI
4 S-II B
5 S-IVB
6 INJEC
7 S-IVB
TRANS
8 FORW
NOSE
9 FORW
AND
10 AFT F
11 AFT F
AND
INITI
12 FORW
TRAJE
13 AFT F
TRAJE
14 PLAN
15 STERI

PLANETARY VEHIC

16 FORW
17 AFT P

MARS ENTRY PHAS

18 FORW
19 AFT P
20 FORW
21 AFT C

ORIBTER PHASE

22 FORW
23 AFT F

4

ANSIT PHASE
E VEHICLE LAUNCH
BURNOUT AND SEPARATION
NGINE BURN AND UPPER NOSE
NG SEPARATION
URNOUT AND SEPARATION
FIRST BURN STARTED
TION INTO EARTH PARKING ORBIT
SECOND BURN — INJECTION INTO
S-MARS TRAJECTORY
ARD PLANETARY VEHICLE SEPARATED AND AFT
FAIRING FOLDED BACK
ARD PLANETARY VEHICLE EQUIPMENT DEPLOYED
CELESTIAL REFERENCE ACQUISITION INITIATED
LANETARY VEHICLE SEPARATION
LANETARY VEHICLE EQUIPMENT DEPLOYED
CELESTIAL REFERENCE ACQUISITION
ATED
ARD PLANETARY VEHICLE INTERPLANETARY
CTORY FIRST CORRECTION
LANETARY VEHICLE INTERPLANETARY
CTORY FIRST CORRECTION
ETARY VEHICLE CRUISE
LIZATION CANISTER SEPARATION

LE ORBIT ACHIEVEMENT PHASE
ARD PLANETARY VEHICLE ORBIT INSERTION
LANETARY VEHICLE ORBIT INSERTION

E
ARD PLANETARY VEHICLE CAPSULE SEPARATION
LANETARY VEHICLE CAPSULE SEPARATION
ARD CAPSULE DEORBIT AND OPERATIONS
APSULE DEORBIT AND OPERATIONS

ARD FLIGHT SPACECRAFT ORBITAL OPERATIONS
LIGHT SPACECRAFT ORBITAL OPERATIONS

Figure 2-1: Voyager Mission Profile

5

Following S-IVB shutdown, the forward Planetary Vehicle is separated and the aft nose fairing is folded back for subsequent separation of the aft vehicle. Upon separation, the external equipment on each vehicle is deployed and maneuvers are initiated to acquire celestial reference. There is a 0.84 cumulative probability of success associated with separating at least one vehicle, attitude reference acquisition, and subsequent positioning on the specified heliocentric transfer trajectory. This probability value includes the trajectory correction maneuvers described below.

Subsequent to Planetary Vehicle separation, the S-IVB with attached aft nose fairing performs a retromaneuver if necessary to satisfy planetary quarantine constraints and separation requirements established by the Spacecraft System.

Approximately 4 days after launch, the two Planetary Vehicles perform midcourse maneuvers to correct velocity or pointing errors imparted by the launch vehicle and to bias the vehicles so that they arrive at Mars at least 10 days apart. If additional mid-course maneuvers are necessary, they will be performed not later than 30 days before Mars encounter. The Flight Capsule sterilization canisters are jettisoned into a non-impact trajectory ten days prior to Mars encounter.

Depending on the specific launch date, the transit time to Mars can range between 120 and 235 days. Measurements of the interplanetary medium and engineering data from the vehicles are transmitted to Earth during this phase.

Planetary Vehicle Orbit Achievement Phase--At Mars encounter, the orbit-insertion engine is fired to insert the vehicle into a nominal orbit having a 1000-kilometer periapsis altitude, a 20,000-kilometer apoapsis altitude, and a 13.8-hour orbital period. One or more orbit-trim maneuvers may be performed to achieve the final specified orbit. The cumulative probability of success is 0.78 as a goal for acceptable orbit placement of at least one Planetary Vehicle.

Mars Entry Phase--Between 3 and 10 days after orbit achievement, the Flight Capsule is separated and deorbited so that it will impact the Martian surface. Throughout descent, data is transmitted from the capsule to the orbiter and stored for later transmission to Earth. Cumulative success values for one Flight Capsule are apportioned as follows: separation of Flight Spacecraft and capsule--0.76; capsule injection into impact trajectory--0.72; capsule performance--0.63.

Orbital Phase--During orbit around Mars, the scientific instruments and cameras on the spacecraft sense the Martian environment. Spacecraft operation is required for one month and as long thereafter as possible. There is a 0.65 cumulative probability of success that at least one Flight Spacecraft will perform the completed mission sequence defined in the preceding paragraphs.

2.1.2 Science Subsystem Characteristics and Requirements

The Science Subsystem, which is government-furnished equipment provided by JPL, will be mounted on the Spacecraft Bus. It consists of science instruments and associated electronics, data automation equipment, planetary scan platforms, instrument booms, power control, electronic switching, and science command decoding equipment.

The Science Subsystem, as identified by JPL, obtains information concerning the atmosphere, surface, body characteristics, and environment of Mars by performing unmanned experiments in orbit about the planet. It also provides the secondary capability to expand knowledge of the interplanetary medium between the orbits of the Earth and Mars by obtaining scientific and engineering measurements while the Planetary Vehicle is in transit.

The Science Subsystem, as described by JPL, is summarized in the following paragraph.

Weight--A preliminary weight allocation for the Science Subsystem is given below.

Components	(Pounds)
Instruments on Scan Platform	100
Instruments on Spacecraft Bus	140
Data Automation Equipment, Booms, Platforms, Cabling, etc.	160
Total Weight	400

Electrical Power--Primary power allocation is a minimum of 70 watts to science instruments on the scan platform and 40 watts to instruments in the Spacecraft Bus. Secondary power to operate electromechanical devices, such as the platform control motor, will be provided as necessary.

Data Automation Equipment (DAE)--D.C. signals will not be carried across the interface of the Spacecraft Bus and data automation equipment. The DAE will furnish a separate data line that includes real-time science data as well as a separate data line including non-real-time science data. The DAE will perform all required input buffering and data formating for the science data recorders and will control their operational sequences.

Platform Pointing--The random angular motion of the scan platform will be limited so that the maximum random angular velocity of the optical pointing axis relative to the planet local vertical is less than 1.3×10^{-4} radian per second. The scan-platform control subsystem will align the boresight axis of the platform to within 1.0 degree of the center of the planet. (This applies only during planetary orbit and when Sun-Spacecraft-planet angle is equal to or less than 80 degrees.) The design of the scan platform and mount will keep the vibration environment of the instruments to within Voyager specification limits.

Thermal Control--The spacecraft and the thermal control system will be designed to maintain all scientific instruments and electronics to within specified temperature limits throughout the entire spacecraft mission.

Science Instruments--The body-fixed science instruments are mounted on the Spacecraft Bus in locations that satisfy look angles and positions. Instrument booms will be necessary to meet requirements of the magnetometer and some particle and photon sensors. Special antennas will also be required for radio-frequency experiments. Sequential deployment of booms and antennas may be required to satisfy instrument functional requirements.

Science Data Management--The Science Subsystem does not store data within its own system; therefore, the Flight Spacecraft will store at least 5×10^7 bits of data, with a design goal of 5×10^8 bits. Individual Science Subsystem tape recorder requirements are for: two television cameras, the IR and UV spectrometer, the IR scanner, science data generated during maneuvers, and fields and particle measurements during a solar flare.

Functional Restraints on Spacecraft Bus--The Science Subsystem imposes functional restraints on the following spacecraft subsystems: power, computing and sequencing, telemetry, command, data storage, structural and mechanical, pyrotechnic, temperature control, guidance and control and cabling.

2.1.3 Flight Spacecraft Subsystems

The requirements of a given subsystem that affect the requirements of other subsystems are listed in this section under the appropriate subsystem headings. Requirements that specifically apply to one subsystem will be listed in Section 4.0 (Volume A).

General requirements that apply to more than one subsystem are:

1) No gas or solid ejecta from any subsystem will contaminate Mars.
2) No single failure mode of an electrical or electronic part or component will cause a catastrophic effect (defined in Table 3.17-1) on the mission.
3) Slip rings will not be used in any subsystem.
4) All subsystems and the bus structure will be capable of withstanding the shock, linear acceleration, and vibration loads resulting from transportation, launch and injection, separation and maneuvers, and orbit insertion.
5) The Flight Spacecraft will be contained within a dynamic envelope having a maximum diameter of 240 inches and a maximum length of 208 inches.
6) Signals from the computing and sequencing subsystem will initiate Flight Capsule separation and emergency separation of the capsule:
   a) Separation of the Flight Spacecraft and Flight Capsule normally will be effected above the structural field joint;

b) An emergency separation system will be designed for a possible malfunction within the Flight Capsule.

7) Separation of the Planetary Vehicle and launch vehicle will be controlled by the launch vehicle.

8) Materials allowable strength information will be taken from MIL-Handbook 5, JPL-approved contractor test data, or other JPL-approved sources.

9) All subsystems will be designed with proven techniques that allow for uncertainties, off-design conditions, and partial failures.

10) All subsystem designs will incorporate as many space-proven materials, connectors, wire insulation, and wire types as possible to improve reliability.

11) The flexibility to revise interconnecting wiring or to add wires as the program progresses will be considered.

12) To suppress corona, rapid venting of entrapped gas and the use of selected materials and techniques will be considered.

13) Engineering measurements of all subsystems will be made to evaluate subsystem operation.

14) The grounding concept of the Flight Spacecraft and its subsystems will be as defined in the September 17, 1965, JPL-released general specification, "Performance and Design Requirements for the Voyager 1971 Spacecraft System, General Specification For, Preliminary." Boeing interprets the ground concept to be such that the subsystem power buses are referenced to a single ground point. Only one of the buses is tied to the spacecraft structure. All signal and rf circuits are referenced to a separate, single point ground. Pyrotechnic circuits are referenced to a third, single point ground.

15) Regulated d.c. power will not be distributed to spacecraft loads.

16) Ground-command override of critical automatic switching will be provided.

17) Flight Spacecraft operations will be automatic and capable of a complete nominal mission without ground command; however, ground command will be provided as a backup.

18) The mass properties of the Flight Capsule will be:

   a) 1971 and 1973 (2000 pounds)--$I_X = I_Y = I_Z =$ 1500 slug ft^2;

   b) 1975 and 1977 (8000 pounds)--$I_X = I_Y = I_Z =$ 8000 slug ft^2.

19) The spacecraft will be transportable by air and/or soft-ride van.

20) Materials will be selected for a 2-year lifetime, on the basis of corrosion resistance, fungus resistance, nonmagnetic properties, structural efficiency, compatibility with thermal control coatings, and vacuum stability.

21) Design will prevent vacuum welding of surfaces requiring relative motion.

2.1.3.1 Power Subsystem

The Flight Spacecraft electrical-power subsystem will contain two power sources. Solar panels will provide power to the Flight Spacecraft as soon as the Sun is acquired after interplanetary trajectory injection. This power will be continuous during solar acquisition until the end of Flight Spacecraft operation. The second source, batteries, will be used during all off-Sun periods.

The power subsystem will provide a precision clock, clock oscillator, and countdown register. The power subsystem will continuously distribute conditioned power to the Flight Spacecraft.

2.1.3.2 Computing and Sequencing Subsystem

The computing and sequencing subsystem will provide all timed, sequenced, or ground-directed commands to all other Flight Spacecraft subsystems. All timed or sequenced commands will be preset into the subsystem before launch, thus allowing the subsystem to execute all functions from launch to mission completion without further ground command. The specific subsystem requirements are:

1) The subsystem will be capable of:
   a) Restoring total state of C&S and its outputs after a power failure;
   b) A timing accuracy of 1 part in 10^6;
   c) Executing 140 different commands;
2) The subsystem will provide timed outputs for cumulative, periodic, and/or reference times from:
   a) Launch--both independent of and dependent on launch date;
   b) Arrival at the planet--both independent and dependent on launch date and/or date of arrival;
   c) The receipt of a radio-initiated command.

2.1.3.3 Guidance and Control Subsystem

The guidance and control subsystem will acquire and maintain external attitude reference, which is necessary to stabilize the Planetary Vehicle attitude. It will orient the vehicle for orbit insertion, midcourse maneuvers, and orbit trim maneuvers, and will control sensor and antenna pointing. This subsystem consists of an attitude reference unit, autopilot, reaction-control assembly, planet-sensor assembly, and antenna-pointing mechanism and controller. Specific requirements of this subsystem are:

1) A passive control stabilization, such as derived rate, will be used during cruise phase;
2) The star sensor in the attitude reference unit will not have moving parts that are subject to wear or cold vacuum welding;
3) The subsystem will make measurements of the planetary approach geometry, which will be used for Mars orbit determination;
4) The reaction control unit will maintain or change the attitude of the Planetary Vehicle during cruise and the attitude of the Planetary Vehicle and Flight Spacecraft during Mars orbit;
5) The reaction control unit will orient vehicle attitude for trajectory corrections, orbit insertion, orbit trim, and capsule-separation maneuvers.

2.1.3.4 Radio Subsystems

The radio subsystem will provide communication between the Flight Spacecraft and the DSN. It provides the means for the Flight Spacecraft to send telemetry data to the DSN and for the DSN to send operation directions to the Flight Spacecraft command system. It will also receive Flight Capsule data signals. The design of the radio subsystem will be compatible with the use of 85-foot and 210-foot DSIF S-band antennas. The following requirements also apply to the radio subsystem:

1) The subsystem will be capable of receiving data at the rate of 5×10^4 bps from the Flight Capsule during the capsule's flight from Mars orbit to the surface of Mars;
2) The subsystem will be capable of simultaneously receiving commands and transmitting both telemetry and tracking signals.

3) The nominal rf power level transmitted from the Flight Spacecraft will be 50 watts--minimum power level, 36 watts;

4) Data rate will be at least 600 bps at the end of 6 months of orbital operations for a 13.8 hour orbit period.

2.1.3.5 Telemetry Subsystem

The telemetry subsystem will accept data outputs from other Flight Spacecraft subsystems and transducers and process this data for input to the radio subsystem as a serial pulse-code-modulated digital signal. The telemetry subsystem will provide an adequate number of different bit rates so that all phases of the flight mission profile are properly covered.

Specific requirements for the telemetry subsystem are:

1) For a single launch, the telemetry data rate will not exceed 15,000 bps;

2) The Flight Spacecraft will be capable of transmitting a minimum of 2.5×10^6 bits per day of cruise science data to Earth;

3) The Flight Spacecraft will be capable of transmitting a minimum of 5×10^7 bits per day of science payload data to Earth at the end of the 6-month orbit operation about Mars;

4) The information bit error rate will not exceed 5 in 10^3;

5) Reduced (degraded) modes of operation will be provided if the primary mode fails;

6) The Flight Capsule will have a one-way communication link with the Flight Spacecraft.

2.1.3.6 Command Subsystem

The command subsystem, in conjunction with the radio subsystem and the DSIF ground equipment, provides a means of controlling Flight Spacecraft functions from Earth by radio command. The command subsystem also sends quantitative commands to the computing and sequencing subsystem to determine maneuver parameters. Specifically, the command subsystem will:

1) Update initiate-times of all commands;
2) Provide commands to C&S and DAE for execution and ground verification;
3) Select modes of operation;
4) Select an emergency or low data rate;
5) Provide capability for ground-command override.

2.1.3.7 Data Storage Subsystem

The data storage subsystem will record digital data from the Spacecraft Science Payload subsystem and will record engineering and cruise science data from the Flight Spacecraft telemetry subsystem. The subsystem will buffer information received from the Flight Capsule relay link and the capsule relay link recorder at rates compatible with the telemetry subsystem. The minimum requirements compatible with telemetry transmission rates and orbital period are:

1) Science data storage capability will be 2.61×10^8 bits, excluding capsule relay link recorder;
2) Total data storage capacity of 3.82×10^8 bits, and a reproduce capability of 7500 bps;
3) Data storage and reproduce requirements for the individual recorders are listed in Table 4.1.4-1.

2.1.3.8 Structures and Mechanical Subsystem

The structures portion of the structures and mechanical subsystem supports all major elements of the Planetary Vehicle including the Flight Capsule, the propulsion module, and the Science Subsystem. The primary structure also serves as a thermal load path between many of the electronic assemblies and provides meteoroid protection for internal components. The mechanisms portion of the subsystem provides for attachment and deployment of the low- and high-gain antennas, solar panels, guidance scan platform, Flight Capsule emergency separation, and Planetary Vehicle separation.

2.1.3.9 Pyrotechnic Subsystem

The pyrotechnic subsystem deploys solar panels, magnetometer, and antennas, and initiates protective-cover release, emergency Flight Capsule separation, Planetary Vehicle separation, solid motor ignition, and propulsion subsystem valve actuations. Pyrotechnic initiation is commanded by the computing and sequencing subsystem (with backup command from the command subsystem), with the exception of solar-panel deployment, which receives initial command from the separation-initiated timer (SIT) with backup from the computing and sequencing subsystem and command subsystem.

The following specific requirements apply to the pyrotechnic subsystem:

1) Power to the pyrotechnic subsystem will be enabled by the normally open contacts of a safe-and-arm device;
2) No single or common failure mode (including procedural deviation) will both arm and command the pyrotechnic subsystem;
3) Power for actuation of electro-explosive devices will be drawn only from electrical energy storage within the pyrotechnic firing equipment.

2.1.3.10 Temperature Control Subsystem

The temperature control subsystem will maintain the Flight Spacecraft within the specified operating temperature range throughout a normal mission, and will provide a reasonable margin for uncertainties and off-design conditions.

The Flight Spacecraft temperature is controlled by radiation-type vacuum insulation blankets, individual temperature-actuated louvers, radiation-shield liners, thermal coatings, low-thermal-conductance joints, and backup electrical heaters.

The temperature control subsystem will be designed to:

1) Depend on radiation as the primary mode of heat dissipation;
2) Operate satisfactorily during Planetary Vehicle maneuvering;
3) Operate satisfactorily throughout variations of electrical heat dissipation in the Flight Spacecraft;
4) Be compatible with meteroid protection requirements.

2.1.3.11 Cabling Subsystem

The Flight Spacecraft cabling will provide electrical interconnection between equipment items in all parts of the Flight Spacecraft as well as the separable items.

The following requirements apply:

1) The cabling subsystem will be designed to supress electromagnetic interference, magnetic interference, corona, and capacitance coupling;
2) The cable assemblies will be separated according to function:
    a) Power circuitry will be routed separately from all other systems,
    b) Pyrotechnic circuit cables will be routed separately from all other circuits and from each other,

c) Signal circuits (including the data system) will be routed separate from a) and b),

d) RF cables will be routed separately where necessary,

e) Separate cable assemblies will be required within the nose fairing for fairing separation circuits.

2.1.3.12 Propulsion Subsystem

The propulsion subsystem will provide the thrust required to: correct the interplanetary trajectory of the Planetary Vehicle to ensure proper Mars encounter, place the Planetary Vehicle into orbit around Mars, and ensure proper Mars-orbit trim.

Because of these functions, the propulsion subsystem will provide a high thrust capability and a restartable low thrust capability.

The propulsion subsystem will be modularized. To the maximum degree possible, consistent with good design, it will be installed as a complete package.

The following are subsystem requirements.

1) The propulsion subsystem will weight no more than 15,000 pounds; the structural system must support this weight.

2) The subsystem must be capable of performing after space vacuum environment storage up to 400 days.

3) Internal components that come in contact with freon and nitrogen will be compatible with sterilization processes defined in JPL Specifications XSO-30275-TST-A and GMO-50198-ETS.

2.1.3.13 Planetary Vehicle Adapter

Each Planetary Vehicle Adapter will secure one Planetary Vehicle to the launch vehicle nose fairing. The adapter will be a simple mechanical system that provides structural support to the Planetary Vehicle during the boost phase, and will be one that is easily removed following injection into a heliocentric trajectory. The adapter will retain the separation system components after separation from the Planetary Vehicle. It will support a Planetary Vehicle weight of 19,500 pounds.

2.2 DESIGN RESTRAINTS

Section 2.2 describes the design restraints applicable to the Flight Spacecraft, with particular emphasis on natural and induced environments. Also included are the restraints imposed by planetary quarantine, vehicle attitude, perturbation and separation requirements, and reliability and safety.

All design concepts, materials, and components considered for the Voyager 1971 mission will have a development freeze date 5 months from Phase IB go-ahead. Only the design concepts that have been developed and have demonstrated feasibility by that date will be considered for inclusion in the Voyager 1971 mission.

2.2.1 Radiation Sources

The flux produced by artificial or natural radioactive material on-board the Flight Spacecraft must not produce an increase in the counting rate of the interplanetary radiation instruments in excess of 1.0 percent of the interplanetary background rate.

2.2.2 Magnetic Interference

Maximum Magnetic Field--The maximum magnetic field caused by the Flight Spacecraft, at the science magnetometer sensor, will be:

1) Less than 1 gamma for a demagnetized Flight Spacecraft under all normal operating conditions;
2) Less than 10 gamma for a Flight Spacecraft exposed to a magnetic field of at least 25 oersteds.

No single assembly will contribute more than 20 percent of the total allowable field at the magnetometer sensor.

Materials--All materials, processes, and parts used in the Flight Spacecraft should be nonmagnetic or, if ferromagnetic, they should be restricted to usage where magnetic field constraints are not violated.

Magnetic Field Stability--Magnetic fields will be stable to the extent that the field at the magnetometer sensor does not change by more than 0.1 gamma from that obtained in the normal cruise condition due to:

1) A change between any two operational modes after initial transients have died out (less than a few minutes);
2) Voltage or current changes over the full design range of the subsystem for any individual operating mode.

The Flight Spacecraft will be magnetically evaluated after being subjected to:

1) Magnetization in a magnetic field of not less than 50 oersteds;
2) Demagnetization by a decreasing alternating field of initial peak value less than 100 oersteds;
3) Operational modes of the energized spacecraft.

In designing Flight Spacecraft circuitry, the requirements for subsequent deperming, Item 2 in preceding paragraph, will be given consideration, and the enclosed area of circuit loops will be held to a minimum.

2.2.3 Environment

This section describes the environment encountered for all parts, components, and the Flight Spacecraft. JPL's "Voyager Environmental Predictions" document is identified as the primary source of the following data. Subparagraphs of this section contain a matrix of environmental parameters considered versus location, event, or operation. The matrix symbols are defined as follows. NA: Not applicable (not a controlling spacecraft design factor): X: indicates that either a partial or complete estimate of the environment is discussed or the JPL "Voyager Environmental Predictions" document description is referenced; a blank indicates that no estimate is currently available.

2.2.3.1 Manufacturing and Shipment Environment

Table 2-1 shows environmental considerations for manufacturing, shipping, and storage.

Table 2-1: ENVIRONMENTAL CONSIDERATIONS
IN MANUFACTURE, SHIPMENT, AND STORAGE

Par. No.	Parameter	Manu-facture	Bench Testing	Potting and Conformal Coating	Repair	Shipping, Hand Carry	Shipping, Commercial	Shipping System	Storage
1.	Temperature					X	X	X	X
2.	Humidity						X	X	X
3.	Shock	X	X	X	X	X			
4.	Vibration			NA					
5.	Contaminants	X	X	X	X				
6.	Solvents and Chemicals					NA	NA	NA	NA
7.	Magnetics	X	X	X	X		X	X	X

Symbols: NA - Not Applicable
X - Environment Specified
Blank - Environment Not Defined

1) Temperature--The temperature of components or assemblies after manufacturing will not exceed flight acceptance testing limits. Temperature of the Flight Spacecraft during shipment and storage will be maintained between 35 and 100^{o}F. Silver-cadmium batteries will be stored between 20 and 50^{o}F at ETR before installation in the Flight Spacecraft.

2) Humidity--The Flight Spacecraft will not be exposed to relative humidity in excess of 50 percent.

3) Shock--Shock caused by accidental drop of unpackaged parts or components during handling for inspection, assembly, tests, or transportation is not expected to exceed:

    a) Two-inch vertical drop onto a wood (fir) bench flat on the face of the component;

    b) Tilt drop--a component edge will be used as a pivot and the opposite edge will be raised 4 inches from the bench, or as high as practical.

4) Vibration--Vibration environment during manufacturing, bench testing, and repair will be controlled. The maximum vibration environment will be defined.

5) Contaminants--Exposure to contaminants will be controlled as specified in Section 3.13.

6) Solvents and Chemicals--During shipment and storage, the Flight Spacecraft will be protected to preclude exposure to chemicals and solvents.

7) Magnetics--In the manufacturing process, the Flight Spacecraft and its components can be exposed to magnetizing fields in addition to that of the normal Earth's field. The effect of such fields will de depend on the amount and type of magnetic material in the Flight

Spacecraft components, the magnitude and proximity of such fields, and mechanical forces exerted on the component. A reduction in any of these reduces the effects. The maximum magnetic field to which Flight Spacecraft components will be exposed during manufacture is 100 oersteds. The magnetic stability of the Flight Spacecraft or Flight Capsule will be affected by shock and vibration during shipping. The exact amount of this effect will be determined.

2.2.3.2 Test and Prelaunch Environment

Table 2-2 shows environmental considerations during test and prelaunch.

Table 2-2: ENVIRONMENTAL CONSIDERATIONS DURING TEST AND PRELAUNCH

Par. No.	Parameter	Subsystem Labs	SAF (S/C Assy Facility)	ESA (Explosive-Safe Area)	On Pad
1.	Temperature		X		X
2.	Humidity		X		
3.	Vibration	NA	NA	NA	
	Pressure				
	Shock				NA
4.	Electrical Transients	NA	X	X	X
5.	Corrosive Atmosphere	NA	NA		
	Contamination				
	ETO	NA			
6.	EMI	X	X	X	X
7.	Magnetics	X	X	X	
	Heat Steriliza-tion				
	Deperming and Mapping				
8.	Explosive Atmosphere				X
	Solar Radiation				
9.	Acceleration				X

Symbols: NA - Not Applicable
X - Environment Specified
Blank - Environment Not Determined

1) Temperature--The temperature of the Flight Spacecraft will be maintained between 35 and 100°F before battery installation and between 35 and 75°F after installation. During prelaunch operations, Flight Spacecraft subsystems will be designed to tolerate local temperatures corresponding to a bulk gas temperature within the nose fairing of between 45 and 65°F in the steady state.

2) Humidity--The Flight Spacecraft will not be exposed to relative humidity in excess of 50 percent.

3) Vibration--The vibration environment of the subsystems through the explosive-safe area (ESA) will be limited by a control program. The limits for vibration exposure will be estimated; however, these limited will not influence the Flight Spacecraft design and, therefore, are indicated in Table 2-2 as not applicable.

4) Electrical Transients--Induced voltage levels up to a 50-volt peak are expected to be coupled into Voyager system cabling due to power transients. A representative transient is shown in Figure 1 of the JPL "Voyager Environmental Predictions" document.

5) Corrosive Atmosphere--The Flight Spacecraft will be protected from exposure to corrosive atmosphere at the subsystem laboratories and spacecraft assembly facility (SAF); therefore, this environment is indicated as not applicable in Table 2-2.

6) Electromagnetic Radiation--At the launch pad, launch vehicle transmitters and Cape Kennedy rf sources, such as radar, will provide an electromagnetic environment to the Flight Spacecraft. No information exists on the shielding properties of the launch vehicle nose fairing, but, if a transparent nose fairing will be the worst case, rf power densities in Table 2-3 will be used.

Table 2-3: POWER DENSITY LEVELS (DBM/METER)

Frequency	ESA	Enroute	Pad	Ascent	Parking
0.150-100	2	2	2	2	--
100-150	4	4	4	4	--
225-260	--	--	36	36	36
400-550	10	10	10	10	--
1200-1400	20	30	30	30	--
2200-2900	--	0	--	--	--
5400-5900	18	28	28	28	23
8500-10,000	20	30	30	30	23

In addition to frequency ranges listed in Table 2-3, very-low-frequency radiation spectra of lightning discharge occur from about 1 to 40 kc.

7) Magnetics--During magnetic stability evaluation and testing, d.c. magnetizing and a.c. demagnetizing fields as high as 100 oersteds will be used on subassemblies, assemblies, and the assembled Flight Spacecraft. The frequency of these demagnetizing fields, which will have an effect in circuit loops, will be 60 cps for assemblies and smaller components but will be restricted to less than 1 cps for system tests. It is assumed that small components and nonelectrical hardware will be exposed to concentrated 60-cps demagnetizing fields as high as 1000 oersteds.

8) Explosive Atmosphere--It is assumed that the Flight Spacecraft will be exposed to an explosive atmosphere when on the launch vehicle. This atmosphere might consist of oxygen, hydrogen, RP-1, and hydrazine.

9) Acceleration--It is assumed that the Flight Spacecraft will be exposed to accelerations in the upward direction of 1.5 g's maximum during hoisting. Maximum incremental accelerations during onsite transportation should not exceed 0.5 g in any direction. While on the launch vehicle before launch, the Flight Spacecraft should not

experience accelerations greater than 0.15 g due to ground-wind excitation of the launch vehicle.

2.2.3.3 Launch Environment

Table 2-4 shows environmental considerations during launch.

Table 2-4: ENVIRONMENTAL CONSIDERATIONS DURING LAUNCH

Paragraph No.	Parameter	Engine Ignition to L + 10 seconds	L + 10 Seconds to Transonic	Transonic	Transonic to SIC Staging	SII Operation	Nose Fairing Separation	SIV-B Operation	Parking Orbit	SIV-B Operation	Planetary Vehicle Separation
1.	Shock		NA	NA			X	X	NA	X	X
2.	Acceleration, Low Frequency	X	X	X	X	X	NA	X	NA	X	NA
2.	Vibration Random	X	X	X	X	X	NA	X	NA	X	NA
3.	Acoustic	X	X	X	X	NA	NA	NA	NA	NA	NA
2.	Static Acceleration	X	X	X	X	X	NA	X	NA	X	NA
4.	Temp. and Thermal Transients	X	X	X	X	X	X	X	X	X	X
5.	Pressure Reduction	NA	X	X	X	X	X	NA	NA	NA	NA
6.	Electromagnetic Radiation	X	X	X	X	X	X	X	X	X	X
7.	Electrical Transients	X			X	X				X	
8.	Electrically Conductive Gas	NA	NA			NA	NA	NA	NA	NA	NA
9.	Electrostatic Chg and Discharge	NA						NA	NA	NA	NA
10.	Cleanliness	X	X	X	X	X	X				
11.	Solar Radiation	NA	NA	NA	NA	NA	X	X	X	X	X
12.	Albedo	NA	NA	NA	NA	NA	X	X	X	X	X
13.	Magnetics	X	X	X	X	X	X	X		X	X

Symbols: NA - Not Applicable
X - Environment Specified
Blank - Environment Not Defined

1) Shock--Local shocks will be experienced by the Flight Spacecraft due to pyrotechnic firings. Shock response from these environments is approximated by an input consisting of 200-g terminal peak saw-tooth with a rise time of 0.7 to 1.0 millisecond. There are transient events to be expected during various portions of the flight. The transient events are insignificant compared to nose fairing and Planetary Vehicle separation shocks.

2) Random Vibration, Low-Frequency Acceleration and Static Acceleration-- The response to the random vibration environment used in the design of components shall be based on the environment specified in the JPL "Voyager Environmental Predictions" document. Limit acceleration factors used for preliminary structural design are shown in the following table for both launch and orbit insertion conditions. As indicated by the footnotes, some of the entries have been adjusted to include conservative dynamic response increments.

Condition	Limit Acceleration		
	Z Axis	X or Y Axis	Torsion (About Z)
Launch	+ 5.6 g	± 2 g*	0
	- 2.8 g*	± 2 g*	0
	0	0	10 rad/sec^2
Orbit Insertion	+ 5 g*	0	0
	- 3.5 g**	0	0

* Dynamic response factors included.

** Applicable only to deployed appendages due to their dynamic response at thrust termination.

3) Acoustic--The external acoustic environment shall be the same as that described in the JPL "Voyager Environmental Predictions" document.

4) Temperature and Thermal Transients:

a) Maximum heat rate from the nose fairing to the Flight Spacecraft will be assumed to be 65 watts/sq. ft., totally enveloping IR heat flux during Earth orbit and early cruise;

b) Maximum heat rate from the nose fairing to the Flight Spacecraft will be assumed to be 40 watts/sq. ft. during Earth orbit and early cruise;

c) Maximum aerodynamic (nose-fairing-off) heat rate will be assumed to be 24.2 watts/sq. ft. during Earth orbit and early cruise.

5) Pressure Reduction During Ascent--The ambient pressure versus time is shown in the JPL "Voyager Environmental Predictions" document.

6) Electromagnetic Radiation Environment--See Table 2-3.

7) Electrical Transients--The electrical transients on the umbilical cable wires during launch are assumed to be those of Figure 1 of the JPL "Voyager Environmental Predictions" document. Possible transients due to separation of launch vehicles are to be determined.

8) Electrically Conductive Gas--An electrically conductive gas from the launch vehicle might envelop the Flight Spacecraft and provide a conductive environment. The actual conductive value will be determined.

9) Electrostatic Charge and Discharge--The space vehicle will generate an electrostatic discharge during launch, the magnitude of which will be determined.

10) Cleanliness--Contamination of the Flight Spacecraft after launch can occur from any one of several possible sources: smoking or outgassing

of materials, loosening of particles and debris from the nose fairing and adapter systems by acoustic or mechanical vibrations, V-band separation pyrotechnics, or a retropropulsion maneuver. If pyrotechnic devices are used to separate the nose fairing, special particulate contamination problems will arise.

11) Solar Electromagnetic Radiation--The Solar spectrum outside the Earth's atmosphere will be assumed to have the shape of the "Johnson Curve" with integrated intensity of 127 watts/sq. ft.

12) Albedo Radiation--With an average terrestial albedo of 0.36, the reflected radiation is 45.2 watts/sq. ft.

13) Magnetics--The combination of vibration and the ambient magnetic field associated with launch will affect the magnetic stability of the Flight Spacecraft. The exact nature of the ambient field at launch is not established; however, it is believed to be slightly different from that of the normal Earth's magnetic field because of the umbilical tower and launch vehicle. Vibration and shock will also affect the magnetic stability of the Planetary Vehicle. The extent of these effects will be determined.

2.2.3.4 Postlaunch Environment

Table 2-5 shows environmental considerations relative to postlaunch.

1) Planetary Upper Atmostheres and Interplanetary Space--The atmosphere and matter density for near-Earth, cruise, flyby, and Mars orbit are the same as shown in the JPL "Voyager Environmental Predictions" document.

2) Solar Thermal Radiation--The solar thermal radiation for near-Earth, cruise, Mars encounter, and Mars orbit is the same as shown in the

Table 2-5: ENVIRONMENTAL CONSIDERATIONS RELATIVE TO POSTLAUNCH

Par. No.	Parameter	Near Earth	Cruise	Flyby	Mars Orbit
1.	Planetary Upper Atmosphere	X	X	X	X
2.	Solar Thermal Radiation	X	X	X	X
3.	Corpuscular Radiation				
	Geomagnetically Trapped Particle Radiation	X	NA	X	X
	Galactic Cosmic Radiation	X	X	X	X
	Radioisotope Thermoelectric Generator Rad.	X	X		
	Solar Cosmic Radiation				
	Solar Flares	X	X	X	X
	Solar Wind	X	X	X	X
	Auroral Radiation	X	NA	NA	NA
4.	Meteoroid Env.	X	X	X	X
5.	Electromagnetic Environment	X	X	X	X
6.	Electromagnetic Radiation				
	External Internal	X	X	X	X
7.	Temperature				
8.	Albedo & Planetary Thermal Radiation	NA	NA		

SYMBOLS: NA Not Applicable
X Environment Defined
Blank Environment Not Defined

JPL "Voyager Environmental Predictions" document.

3) Corpuscular Radiation--The geomagnetically trapped radiation, galactic cosmic radiation, radioisotope thermoelectric generator radiation, solar cosmic radiation, and auroral radiation environments are the same as described in the JPL "Voyager Environmental Predictions" document, and the November 12, 1965, "Errata."

4) Meteoroid Environment-- The meteoroid environment to which the Flight Spacecraft will be exposed is the same as that shown in the JPL "Voyager Environmental Predictions" document, and the November 12, 1965, "Errata."

5) Magnetic Environment--The magnetic environment for near-Earth magnetic field and cruise magnetic field is the same as shown in the JPL "Voyager Environmental Predictions" document.

6) Electromagnetic Radiation:
   a) Internal Electromagnetic Radiation--The harmonics from any power subsystem square-wave power supply can be expected to be present on the wiring and to be radiated throughout the Flight Spacecraft;
   b) Transients with a general waveform as shown in Figure 1 of the JPL "Voyager Environmental Predictions" document can be expected to occur on some of the Flight Spacecraft cabling wires. The degree of coupling to adjacent lines can be estimated to be unity as a worst case.

2.2.3.5 Postlaunch Pyrotechnic and Propulsion Events

Table 2-6 shows environmental considerations during postlaunch pyrotechnic and propulsion events.

1) Acceleration--Maximum sustained acceleration in the thrust direction

direction during Planetary Vehicle retropropulsion firing is 3.5 g's (2000-pound capsule).

2) Shock--Ignition of the Flight Spacecraft retropropulsion will generate shock loads within the Flight Spacecraft. Because the ignition transient is unknown at this time, the shock effect on the Flight Spacecraft will be taken as that which would be generated by a step application of thrust.

Table 2-6: ENVIRONMENTAL CONSIDERATIONS DURING POSTLAUNCH PYROTECHNIC AND PROPULSION EVENTS

Par. No.	Parameter	Midcourse Maneuvers	Orbit Insertion	Orbit Trim	Flight Spacecraft Trajectory Deflection	Sterilization Canister Opening	Capsule Spacecraft Separation
	Vibration						
1.	Acceleration		X	X			
2.	Shock	X	X				
3.	Thermal Transients	X	X	X	X		
4.	Charge Buildup (From Combustion)	X	X	X	X	NA	NA
4.	Exhaust Gases	X	X	X	X	NA	NA
	Contamination Microbial Particulate	NA	NA	NA			

SYMBOLS: NA Not Applicable
X Environment Specified
Blank Environment Not Determined

3) Thermal Transients--Thermal transients during maneuvers as severe as going full-Sun to full-shade, and vice versa, can be anticipated.

4) Charge Buildup from Combustion and Exhaust Gases--These environments are the same as shown in the JPL "Voyager Environmental Predictions" document.

2.2.3.6 Planetary Environments

An orbiting Flight Spacecraft will be affected by the albedo, infrared radiation, and surface temperature of Mars.

The albedo of Mars is 0.15. The color index of Mars (1.3 to 1.0) is relatively high compared with the 0.5 index of the Sun.

The yearly mean blackbody effective temperature is assumed to be -13°C (9°F) for the sunlit side and -73°C (-100°F) for the dark side. Variations caused by the eccentricity of the Martian orbit are accounted for by multiplying the yearly mean absolute temperature by the square root of $1.0087 + 0.09405 \cos \theta$, where θ is the heliocentric longitude of Mars from its perihelion.

The surface temperature will range from -98°C (-145°F) for nighttime winter temperatures in the polar regions to +37°C (+98°F) for the maximum daytime ground temperatures for a dark area near the equator during perihelion passage.

2.2.4 Attitude Control Restraints

The elapsed time from space vehicle launch until solar acquisition by the aft Planetary Vehicle will not exceed 2.5 hours, which includes a 90-minute maximum hold time in Earth parking orbit.

The celestial reference will be the Sun and Canopus. The amplitude of the Flight Spacecraft attitude control limit cycle must be held to less than ±0.4 degree along each axis to maintain overall pointing accuracy of the high-gain antenna within 0.9 degree. This restraint can be relaxed during the cruise portion of the mission to the 0.6-degree attitude control limit cycle.

The maximum time duration of first midcourse maneuver will not exceed 65 minutes. This time includes changing the attitude of the Planetary Vehicle to the required maneuver attitude, motor burn, and return of the vehicle to the cruise attitude. During this maneuver, the solar panels will not be off-Sun for more than 50 minutes.

Attitude control pointing angle restraints imposed by trajectory corrections, orbit insertion, and orbit trim are defined in Section 2.3. Flight Spacecraft attitude control will not be impaired when the Sun is occulted by Mars for up to 3.7 hours or when Canopus is occulted by Mars for up to 2.0 hours. The pointing angle error between the Flight Capsule deorbit motor thrust axis and the Flight Spacecraft primary reference axis before capsule separation will not exceed 20 milliradians (3σ) about each axis. The attitude control system must: (1) limit random angular motion of the platform to less than 1.3×10^{-4} rad/sec for the photoimaging experiment; and (2) align the boresight axis of the platform to within ± 1 degree of the center of Mars.

The attitude reference subsystem will not contribute more than a 0.1-degree error to the total allowable pointing error of 0.2 degree when the planet tracker is in operation.

2.2.5 Flight Perturbation Restraints

The following operations will not exceed the capabilities of the reaction control subsystem:

1) Booster separation
2) Propulsion subsystem shutdown transients
3) Separation of the sterilization canister
4) Separation of the capsule
5) ΔV applications

2.2.6 Restraints on Separation of Flight Spacecraft and Flight Capsule

The maximum timing error in the Flight Capsule release signal will not exceed ± 1 minute (3σ).

The maximum ΔV imparted to the Flight Spacecraft by Flight Capsule release will not exceed 0.5 fps.

The timing error allowances on Flight Capsule deorbit after Flight Capsule release are 0.3 minute for a 2000-km circular orbit, and 0.5 minute for latus rectum departure from an elliptical orbit of 1000 km periapsis and 20,000 km apoapsis.

2.2.7 Reliability

Flight Spacecraft reliability design restraints in the form of requirements, guidelines, and preferences are summarized below. Details and application of these restraints, methodology for determining design compliance, and an evaluation of the design response are given in Sections 3.17 and 4.0.

Single Failure--No single failure mode of an electronic or electrical part or component will cause a catastrophic effect on the mission. (See Table 3.17-1 for definition of catastrophic effect.)

Sensing and Switching--Where malfunction sensing and switching is required, the following order of selection governs:

1) Automatic, fail-safe, on-board circuitry;

2) Detection via telemetry; function switch via ground command.

Redundancy Preference--The following order will apply:

1) Cooperative multichannel;

2) Alternate path or functional;

3) Block or standby.

2.2.8 Safety

Safety will receive major consideration in the design of the Voyager Spacecraft System. All safety requirements in "Performance and Design Requirements for the Voyager 1971 Spacecraft System, General Specification for," dated September 17, 1965, will be met. AFETR M-127-1, "Range Safety Manual," will apply, The requirements of AFSCM 80-7, "Handbook of Instructions for Aerospace Vehicle Equipment Design," will be used as guides.

Probability studies will be conducted in Phase IB to establish destruct criteria. However, for Task B, provisions will be incorporated in the adapter for inclusion of a destruct system.

Electro-explosive devices associated with the solid-motor ignition system must be installed in the explosive safe area because encapsulation of the Planetary Vehicle prevents installation on the launch pad. Accordingly, the safe-and-arm system will incorporate a mechanical locking pin to protect against premature ignition of the solid motors prior to launch.

Pressure vessels that will be in proximity to personnel will be designed with a hazard factor of 1.76 and an ultimate load of 2.2 times the design pressure. Pressure vessels and separation components will be capable of withstanding ultimate loads without failure. Rocket-motor cases will have a hazard factor of 1.0. Pyrotechnic devices will not be used when equally reliable mechanical or electromechanical devices are available. When pyrotechnic devices are used, adequate safing devices will be used to prevent premature or unscheduled activation.

2.3 GUIDANCE AND NAVIGATION--MANEUVER ACCURACY AND PROPULSION REQUIREMENTS

The Flight Spacecraft must perform interplanetary trajectory corrections to meet the high degree of precision required in placing a Flight Space craft in a specified Mars orbit. Control accuracies and velocity increments, and trajectory and orbit accuracy requirements imposed on the Spacecraft System are presented in this section.

2.3.1 System Velocity Control Accuracy Requirements

To achieve the trajectory and orbit parameter control accuracy specified in the following paragraphs, the gross control accuracies for maneuvers must be within the following limits, with a 3σ tolerance.

Maneuver Restraints	Pointing Errors	Resolution Errors
For Maneuvers $>$ 1 meters/sec	1.8°	0.1 meter/sec
For Maneuvers $<$ 1 meters/sec	3.0°	10% of ΔV
For Orbit Insertion	----	18 meters/sec

In addition, the ΔV proportional error must be less than or equal to 1.5 percent of ΔV.

2.3.2 Trans-Mars Trajectory Accuracy Requirements

The contribution to Mars-encounter parameter due to control of maneuvers, with no aim-point or flight-time biasing, will not exceed the following (with a 3σ tolerance) after no more than two corrections: semimajor axis in $\bar{R}\ \bar{T}$ plane $\leq$ 100 km; flight time $\leq$ 1.0 minute.

Allowing for orbit determination and aim-point biasing and the possibility of performing three corrections, the encounter accuracies can be controlled to the following 3σ tolerance values: semimajor axis in $\bar{R}$-$\bar{T}$ plane $\leq$350 km; flight times$\leq$2.0 minutes.

2.3.3 Trajectory Maneuver ΔV Requirements

Two or more trajectory correction maneuvers will probably be required. The corrections will include aim-point biasing, flight-time biasing, and correction of deviations due to trans-Mars injection and previous maneuvers. The first maneuver will be executed approximately 4 days after launch, and a second, if necessary, between 25 and 35 days after launch. A third correction, if required, will be for vernier control of encounter and will be executed between 30 and 40 days prior to encounter.

The ΔV required per vehicle to achieve this correction capability is 200 meters per second. ΔV allotment to individual requirements will be made during the actual mission. However, a general premission allotment can be made as follows.

Requirement	ΔV (Meters/Sec)
Flight-Time Adjustment	150
Aim-Point Biasing	10
Correction of Random Errors	40,4σ *

* To account for nonabortive but off-nominal injection errors.

2.3.4 Orbit-Insertion Accuracy Requirements

The orbit parameters at insertion must be controlled, relative to the desired orbit parameters, to within the following accuracies--with a

3σ tolerance: periapsis altitude, 30 percent; inclination, 5 degrees; line of apsides, 10 degrees; period, 10 percent.

2.3.5 Orbit-Insertion ΔV Requirements

Various maneuver techniques are discussed in detail in Section 3.1.3. The Spacecraft Propulsion Subsystem must be able to perform a variety of orbit insertions having several approach velocities. The excess hyperbolic velocity (VHP) requirements depend on the particular combination of launch and arrival dates selected for the launch period. VHP will vary between 2.82 and 4.5 kilometers per second over the allowable range of transit trajectories. The requirement of entering various orbits is manifested by orientations of the line of apsides as much as 90 degrees from the direction of least ΔV and by various required orbit sizes.

The periapsis altitude can be selected on the basis of two alternatives: better observation of the planet surface at low altitudes, and longer lifetime at high altitudes due to reduced atmospheric drag. The apoapsis altitude can be selected on the basis of desired ground coverage, communication to Earth and capsule, solar and Canopus interference by Mars, and data transmission requirements. Until the mission objectives have been clearly defined and a priority established for the scientific experiments, a broad range of orbit altitudes and orientations must be considered.

2.3.6 Orbit-Trim Accuracy Requirements

It is desired that the periapsis altitude, inclination, line of apsides, and orbit period be controlled by performing orbit-trim maneuvers. It is also desirable that this trim be accomplished, with 3σ tolerance, to the following accuracies relative to the desired orbit: periapsis altitude, 1.0 percent; inclination, 1.0 degree; line of apsides (ignoring precessions), 1.5 degrees; period, 1.0 percent.

2.3.7 Orbit-Trim ΔV Requirements

The vehicle must be capable of executing a 100-meter-per-second velocity increment to the Planetary Vehicle while in the Mars-bound orbit.

2.3.8 Mars-Entry Accuracy Requirements

It is desired that the Flight Capsule entry parameters be controlled at capsule deorbit to the following accuracy, with a 3σ tolerance, relative to a specified entry: position of landing, 500 km (with a 300 km goal); entry angle, 6.0 degrees.

2.3.9 Mars-Entry ΔV Requirements

The capsule propulsion subsystem must provide a velocity impulse to establish a new orbit (different from the Mars satellite orbit) that will enter the atmosphere. This trajectory must enter the atmosphere with moderately low entry angles, increasing the time available for communication between the end of transmission blackout and impact and reducing impact velocity. Errors in the landing-site location are sensitive to the deflection errors for these low entry angles; consequently, consideration of allowable landing site dispersions is important in designing the entry trajectory. The ΔV direction and magnitude applied to the capsule varies, depending on where in the orbit the maneuver is performed. The ΔV requirements for the deflection maneuver as a function of the entry velocity, flight path angle at entry, and location in the orbit are discussed preliminarily in Section 3.1.4.

Flight Capsule Entry Trajectory--To meet the simultaneous requirements of a specific landing site and entry within view of Goldstone, and to allow the possibility for independent adjustment of landing latitude, it is recommended that thrust-out of the satellite orbit plane be considered. JPL supplementary information on the entry capsule indicates that a ΔV of approximately 550 meters per second is adequate. The pointing direction of the ΔV affects spacecraft design, but the magnitude of the ΔV does not. The interactions should be investigated jointly as an interface between the capsule contractor and Flight Spacecraft contractor.

2.4 AIMING-POINT SELECTION

The aiming point is defined as the point at which the hyperbolic approach asymptote pierces the $\overline{R}\overline{T}$ plane. This aiming point is described in terms of the conventional RST coordinate system, which has Mars' center as its origin. The aiming point has polar coordinates (B,θ)--where B is the radial distance of the aiming point from the center of Mars, and θ is the polar angle measured positively from the $\overline{T}$ axis into the $\overline{R}$ axis.

To specify a preferable θ, it is necessary to consider the following: (1) occultation of Sun, Earth, and Canopus during the approach hyperbola and in the initial orbit, (2) inclination of the desired orbit, (3) location of the periapsis point for the approach hyperbola and the initial orbit, and (4) probability of impacting Phobos and Deimos.

To specify a preferable B, it is necessary to consider the following: (1) probability of impacting Mars, (2) probability of impacting Phobos and Deimos during the hyperbolic approach, and (3) the desired orbit about Mars.

2.4.1 Aiming-Point Objectives

Trajectory design philosophy is based on maintaining a high probability of mission success while obtaining the best possible conditions to satisfy the scientific objectives. Estimates of the requirements from the scientific objectives are outlined in Section 4.4 of Volume A (D2-82709-6). The B vector influence mission success because aiming-point selection for the transit phase of flight is biased to maintain an overall mission probability of contaminating the planet of less than 10^{-4}. The probability of impacting and contaminating Mars increases as B is decreased. The magnitude of the B vector slightly affects the probability of impacting Phobos and Deimos during the hyperbolic approach trajectory.

At each midcourse correction except the last, the aiming point is moved so that B is controlled by the knowledge of expected spacecraft position. At the last midcourse correction, the B vector is aimed for the minimum allowable B vector for planetary quarantine, or the minimum desired B from a scientific mission standpoint, whichever is the higher. The long-term orbit lifetime and the B-vector dispersions are the main factors in selecting a minimum allowable aiming point. Low B vectors are usually preferred because of the scientific experiment considerations and because the ΔV requirements for orbit insertion decrease as the B vector is decreased. Some of the scientific experiments, as well as operational requirements such as communications, may lead to a preference for larger apoapsis altitudes; this places a secondary constraint on the aiming point because occultation and communication are influenced.

The aiming angle, θ , influences the scientific experiment objectives because it is directly responsible for the resulting inclination of the satellite orbit with respect to the Mars equator. This inclination affects the latitudes of photographic coverage, illumination angles, sensor interference, and interference with lines of sight to the Earth, Sun, and Canopus. The latter two are generally undesirable. The lower inclinations also have higher probabilities of impacting Phobos and Deimos. The effects of orbit inclination are discussed in more detail in Section 3.1.3 and Section 4.4.

The location of the periapsis point of the approach hyperbola is a function of both B and θ. In turn, the periapsis point of the spacecraft orbit is also a function of B, θ, and placement of the orbit periapsis with respect to the hyperbola periapsis.

2.4.2 Selected Aiming Points for an Example Mission

The aiming points for injection into the trans-Mars trajectory and for the three midcourse corrections are placed far enough away from Mars so that the total probability of impacting Mars on the approach hyperbola is less than 0.2×10^{-5}. Because of the highly elliptic nature of the encounter position sensitivity to errors at trans-Mars injection, an initial aiming point, θ, of 116 degrees results in the smallest B-vector magnitude that satisfies the impact probability constraint. For the midcourse corrections, $\theta = 25$ degrees is selected because it ensures freedom from occultation of Sun, Earth, and Canopus along the approach hyperbola. $\theta = 25$ degrees also results in an orbit inclined 46.6 degrees to the Mars equator for a launch of May 1 and an arrival of November 1, 1971.

Assuming a probability of success of 0.9995 for each midcourse maneuver and a probability of success of 0.9990 for orbit insertion at Mars, the B-vector magnitudes (B) for the aiming points of injection and the three midcourse correction maneuvers are, respectively, 11,733 km, 21,307 km, 6,775 km, and 6,436 km. These values were computed by integrating a two-dimensional normal distribution of error over the capture area of Mars in the $\overline{R}\overline{T}$ plane.

For a typical example orbit with a periapsis altitude of 1000 km, an apoapsis altitude of 20,000 km and an inclination of 46.6 degrees with the Mars' equator, the probability of impacting Phobos or Deimos on the approach hyperbola and in the initial Mars orbit is less than 1×10^{-20}. The probabilities of impacting either Phobos or Deimos within 50 years are 2.75×10^{-6} and 1.59×10^{-6}, respectively.

2.5 PLANETARY QUARANTINE REQUIREMENTS

2.5.1 Summary

The Boeing Phase IA, Task A, study concluded that compliance with the planetary-quarantine constraint would require sterilization of the propulsion and attitude control subsystems to prevent contamination by exhaust products. It also concluded that the microbiological burden on the spacecraft surfaces might require reduction by either heat treatment or ethylene oxide because of the probability of contamination of Mars by ejecta resulting from meteoroid impacts on the spacecraft. During Phase IA, Task B, a study was performed to reevaluate and outline the problems within the concept of a single Saturn V launch vehicle per opportunity and two Planetary Vehicles per launch.

The approach used in the Phase IA, Task B, study was to develop system requirements for compliance with the overall mission planetary-quarantine constraint by an analytical procedure similar to that reported in Boeing Document D2-82709-2, "Voyager Spacecraft System Final Technical Report, Volume B."

The results of this analysis indicated that the most critical contaminating events, and those most difficult to evaluate for absolute probability of occurrence, are the events associated with spacecraft propulsion and attitude control subsystem exhausts and meteoroid ejecta.

The analysis to determine the probability of contamination by spacecraft ejecta from meteoroid impacts involved a logical process of calculating subevent probabilities, taking into account such factors and influences as distribution of ejecta; capture by the Martian gravity; particle size, mass, and velocity distributions; effects of atmosphere entry heating; and solar radiation on microbial particles.

The result of this meteoroid-ejecta analysis is presented in Section 2.5.4. The data indicated that the survival probability of micro-organisms ejected from the spacecraft are on the order of 1×10^{-5}. Therefore, it would appear that decontamination of the exterior surfaces may not be required. However, more study is necessary to confirm this position.

Tests were performed during the Task B study of the sporicidal effects of the candidate propellants for the Spacecraft Propulsion Subsystem. The results indicate that hydrazine has significant sporicidal properties.

The solid propellant (aluminized polybutadiene) apparently has little or no sporicidal properties. Firing tests were performed during Task B on a small-scale solid-propellant engine using aluminized polybutadiene to evaluate the survival of micro-organisms in the exhaust. Results indicated a low probability of survival.

Until further study substantiates preliminary findings of the Task B study, the planetary-quarantine requirements on the spacecraft are as follows: (1) sterilization or decontamination of the orbit-insertion TVC fluids and associated hardware interior, (2) sterilization or decontamination of the internal surfaces of the reaction control system, including the propellant, (3) biasing of trajectory aim points to preclude accidental impact at encounter, and (4) the selection of orbit parameters compatible with a probability of impact within 50 years of 1×10^{-5} or less.

2.5.2 Planetary Quarantine Allocation of Event Probabilities

Contamination of Mars can occur in a number of ways. To demonstrate that the mission meets the contamination constraint, these eventualities were identified and then translated into a probability framework. A planetary-quarantine probability analysis was performed to develop a contamination-probability framework for the 1971 Voyager mission so that:

1) Allotments of the total-constraint probability could be intelligently made;
2) Studies of specific contaminating events could be properly integrated;
3) A framework would exist to assist in mission optimization under the contamination constraint.

Probability Allocations--A functional analysis similar to that in Task A was performed to identify mission events that could result in contamination of Mars. Event allocations are shown in Figure 2-2.

To quantify allotments of the constraint probabilities in Figure 2-2, the probability of each of the events identified in the functional analysis must be determined. The probability of each subevent is a function of the equipment reliability, trajectory parameters, structural characteristics of the hardware, ejecta of particles, and the environmental characteristics. In some events the probability is a function of one or more of these.

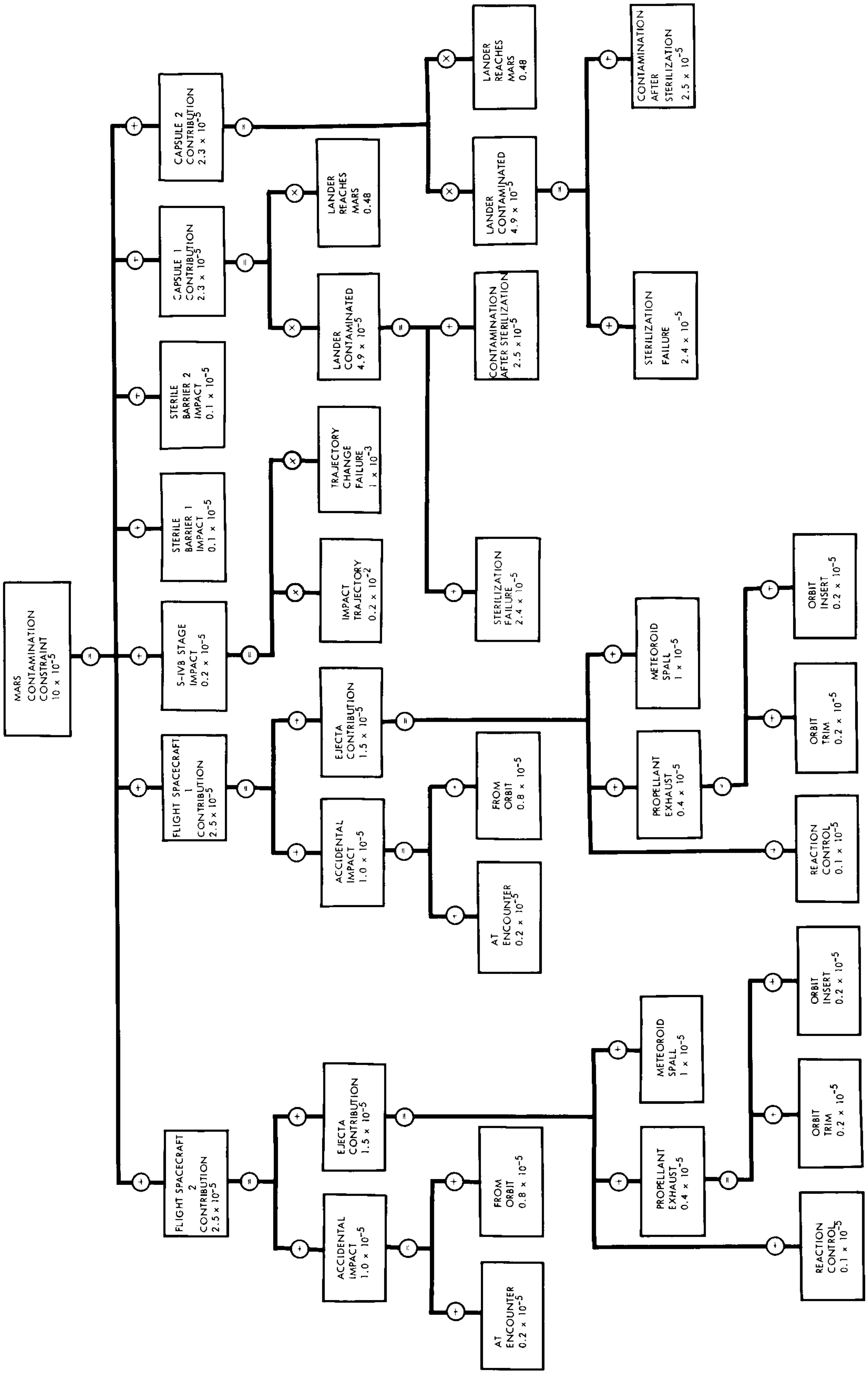


Figure 2-2: Contamination Probability Allocation

For Voyager systems, elements other than the spacecraft were found to be critical, and must be analyzed after detailed design.

Because available time and system and subsystem data did not permit rigorous evaluation of an optimum relative apportionment, the values shown in Figure 2-2 were based on preliminary analysis and engineering judgment. These values, then, are the requirements necessary to satisfy the planetary-quarantine mission constraint.

2.5.3 System Impact Probability

2.5.3.1 Spacecraft Impact

Trajectory-aim-point biasing meets the constraint on probability of spacecraft impact on Mars. During the mission, the spacecraft will be tracked by cooperative effort between its own equipment and Earth-based deep-space instrumentation facilities. The trajectory outcome of each trajectory-control maneuver is determined on Earth with a known certainty, and prepared parametric data is then used to assist in choosing a subsequent aim point, if one is needed.

Finally, the orbit-insertion command will not be given unless the resulting orbit satisfies the impact-probability allocation of 1×10^{-5} for each spacecraft. Discussions of aim-point biasing and orbit parameters are presented in Sections 2.0 and 3.0.

2.5.3.2 S-IVB Nose Fairing Impact

The nose fairing covering the forward Planetary Vehicle module will be jettisoned at 350,000 feet, thus precluding Mars contamination.

The nose fairing covering the rear Planetary Vehicle will remain attached to the S-IVB. Two factors, direction and time of thrust of the S-IVB, enter into the probability that the S-IVB/nose fairing may end up on a trajectory that will impact Mars. Reliability of the subsystems involved in control of these factors is unknown at this time. The capability to direct the S-IVB/nose fairing on a Mars nonimpact trajectory is required in order to meet the probability allocation of 0.2×10^{-5}.

2.5.4 Meteoroid Impact Ejecta Considerations

Probability of planetary contamination by meteoroid impacts on the spacecraft, and by subsequent ejecta depositing viable organisms on the planet's surface was determined in the same manner as in Task A (see D2-82709-2, Volume B, Section 3.3). The general equation for this function is:

$$P_c = P_o \left[1-\exp\left(-\sum_{j=1}^{n} r_j \, P_2^j \, P_3^j \right)\right]$$

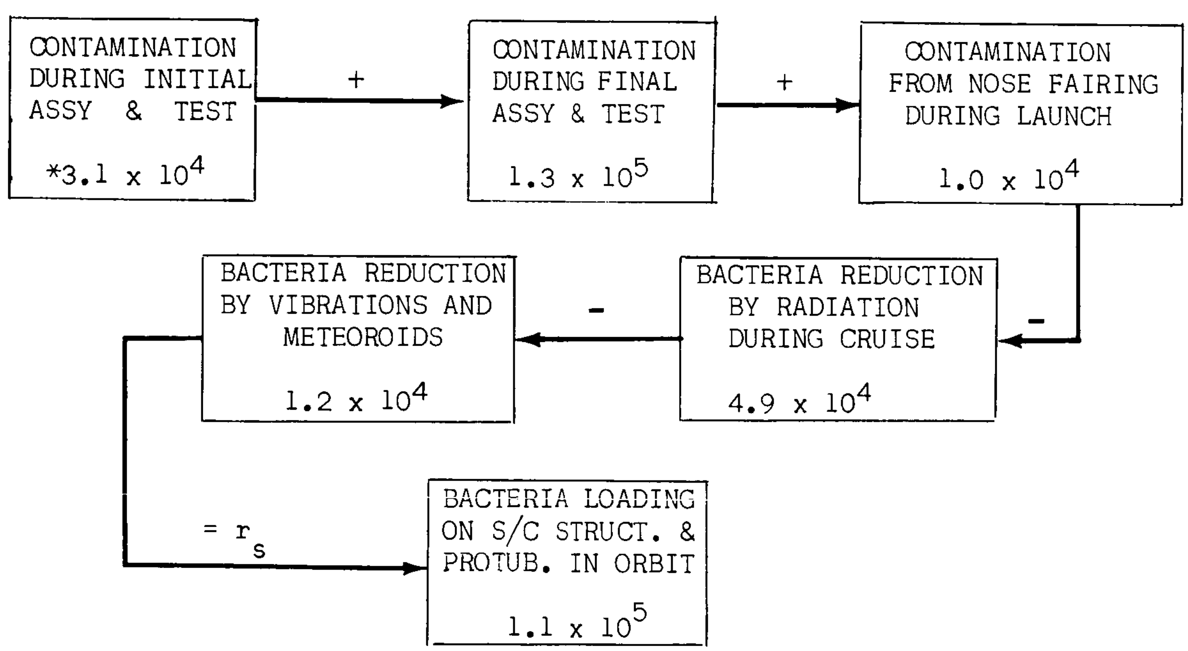
where:

- P_o = Probability of successfully achieving orbit = 0.65.
- r_j = Expected number of spores on exposed surfaces.
- P_2^j = Conditional probability of ejecting an individual live spore from the surface by meteoroid impacts.
- P_3^j = Product of two probabilities: (P_N) the conditional probability of survival of a single ejected spore (alone or on a particle) during capture, entry, and landing; (P_G) the probability that a spore which does survive the ejection, entry, and landing will grow or propagate in the surface environment.

The rP_2P_3 terms in the equation apply to different parts of the orbiting hardware, where j takes the subscript or superscript assigned to each part.

For this analysis, the spacecraft surfaces that will be susceptible to meteoroid impacts were divided into: solar panels (p); antennas (a); spacecraft structures and appendages (s).

Values of r for the various surfaces (solar panels, antennas, and spacecraft and appendages) were derived from estimates of contamination from clean-room fallout, handling during final assembly and test, and from possible contaminant fallout from the spacecraft nose fairing during launch. Estimates of reductions were made from exposure to solar radiation and dislodgment by meteoroid impacts or spacecraft vibrations during the launch and interplanetary cruise. Calculations of r are exemplified in Figure 2-3.



*The bacterial loading is given in number of spores. These were assumed to constitute approximately 10 percent of the total microorganism load.

Figure 2-3: DERIVATION OF r_s (EXPECTED NUMBER OF SPORES ON EXPOSED SPACECRAFT STRUCTURES AND APPENDAGES)

Numerous investigations have indicated that ultraviolet radiation, principally that with wavelengths between 2000 Å and 2700 Å, is effective in killing bacteria. In the spectral band most effective (2500 Å to 2600 Å) incident energy of 1×10^4 to 12×10^4 ergs/cm^2 is required to kill 90 percent of a given microbial population. The smaller value (1×10^4 ergs/cm^2) is effective with a variety of vegetative forms while the higher value of 12×10^4 ergs/cm^2 is necessary for lethal action on bacterial spores (B. subtilis). Under laboratory conditions, the probability of survival would be extremely remote for organisms exposed to the predicted incident ultraviolet radiation in the 2500 Å to 2600 Å band. However, laboratory conditions will not prevail on the Voyager spacecraft.

Variations in microbial sensitivity to ultraviolet radiation have been observed under varying temperatures and under subsequent or coincidental exposure of the test samples to visible light. This phenomenon, photoreactivation, can reduce the lethal efficiency of ultraviolet, by one or more orders of magnitude for certain types of microorganisms. Most of these conditions should be evaluated further by laboratory experimentation.

In an experiment performed during Task B, B. subtilis spores were coated on metal strips and solar panels which were then exposed to UV irradiation of 0.714 watts/ft^2 for periods of one, two and four hours while the chamber was maintained at a pressure of 0.5-mm Hg. White light was used during and immediately following UV exposure. Several panels were subjected to simulated meteoroid impact. The ejecta were collected and exposed to UV radiation for an additional two hours. Bacteriological analyses were then performed on all samples.

A few microorganisms survived UV exposure before impact; none were obtained on the ejecta from the simulated meteoroid impacts. Based on these test results, and because of the uncertainty of the shadowing effects by multiple layers or clumps of spores, the spore reduction on exposed-spacecraft surfaces due to solar ultraviolet radiation was assumed to be 97 percent.

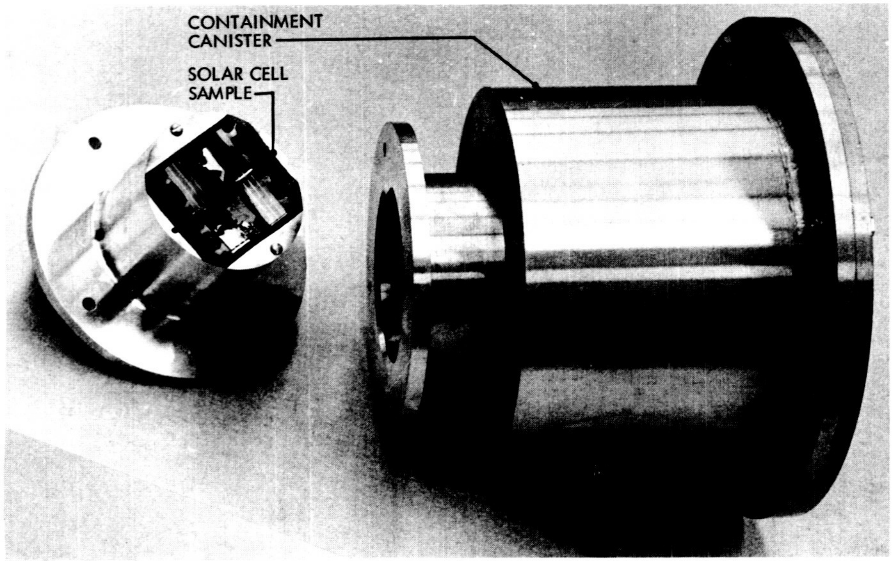
The probability of ejecting spore-carrying material from the spacecraft (P_2) is a function of the exposed spacecraft contaminated-surface areas, materials used in the exposed surfaces, meteoroid flux, exposure time, and the meteoroid impact phenomena related to the ejection of materials and spores. First approximations of the P_2 probability can be made by defining the function as:

$$P_2 = \frac{\text{Contaminated Surface Area Ejected } (A_e)}{\text{Contaminated Surface Area } (A)}$$

With this approximation, P_2 values were calculated, using the meteoroid flux in the vicinity of Mars as contained in the JPL "Preliminary Voyager Environmental Predictions" document, dated September 17, 1965. The values of A_e are based upon experimental studies and analysis performed during Task A and reported in Volume A, D2-82709-1, and in D2-82733-1. The A_e values used in the analysis were: solar panels, 112 cm^2; antennas, 10 cm^2; and spacecraft structures and appendages, 34 cm^2.

Tests were conducted during Task B using the Boeing light-gas-gun facility to simulate typical meteoroid impacts on types of spacecraft structure and other exposed surfaces (see Figure 2-4). These tests indicated that microorganisms will survive in the ejecta from meteoroid impacts.

A series of 14 tests was performed using 10^{-4} gram polycarbonate resin (Lexan) projectiles at velocities approximating 20,000 feet per second



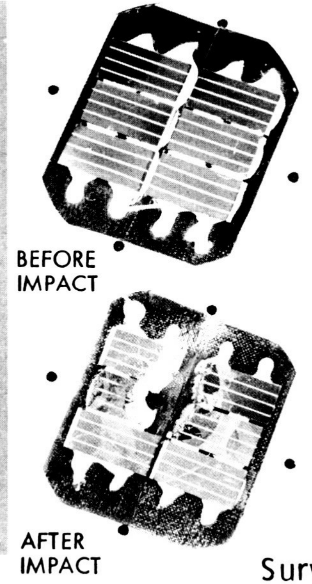


Figure 2-4: Meteoroid Test Samples and Containment Canister

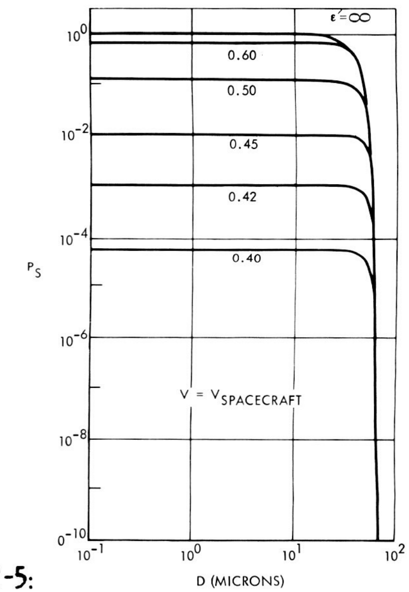


Figure 2-5:
Survival Probability of Spores Ejected at 1000 km Altitude Periapsis

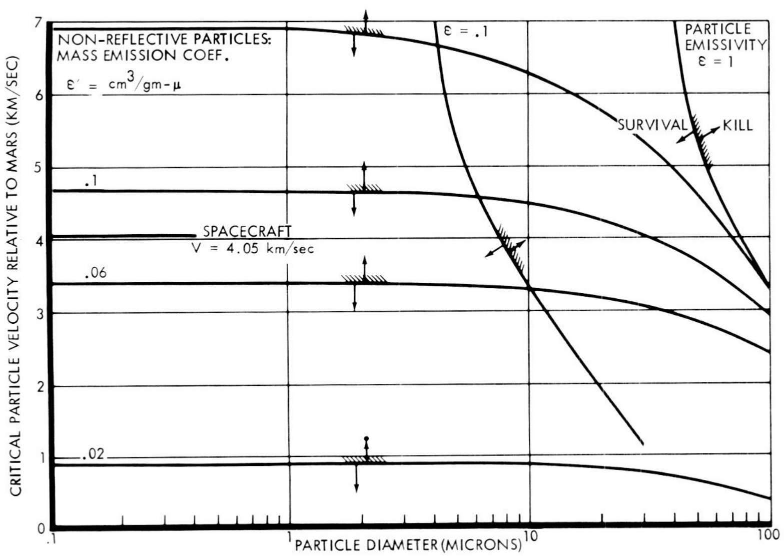


Figure 2-6: Critical Particle Velocity at 1000 km Nominal Periapsis,

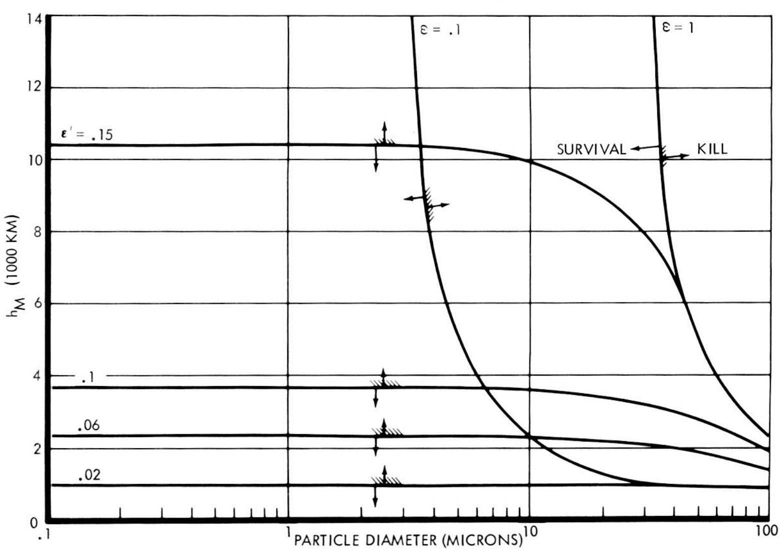


Figure 2-7: Critical Altitude for Aerodynamic Sterilization

striking solar panel and aluminum plate samples. Microorganisms were recovered from all experimental firings.

Using the predicted quantities of spalled ejecta (enumerated previously as A_e values), P_2 values were calculated. They are summarized in Table 2-7 along with r values for the various contaminated surfaces of the spacecraft exposed to possible meteoroid impact.

Table 2-7: VALUES OF r AND P_2 FOR MARS-ORBITING SPACECRAFT--50 YEARS IN ORBIT

Exposed Spacecraft Surfaces	r_j Values	P_2^j Values
External (Solar Cells)	5.0×10^3 Spores	2.4×10^{-4}
Solar Panels		
Internal Laminations	9.8×10^4	2.6×10^{-5}
External (Metal Backing)	4.0×10^3	5.3×10^{-5}
Antenna	1.4×10^3	1.1×10^{-4}
Spacecraft Structures and Protuberances	1.0×10^3	1.2×10^{-4}

In the general equation, P_3 is the product of P_N and P_G. P_N is the product of P_I and P_{II}. P_I is the fraction of all ejected spores alone or on particles that will survive aerodynamic entry heating. P_I is an integral over an abstract space consisting of all possible ejecta-particle initial conditions, sizes, and materials. The integrand is the number distribution of spores times the conditional probability P_S that an individual spore on a specific trajectory will survive aerodynamic entry heating. The distribution function of speed and mass is at present undefined.

Analysis based on recent NASA thermal-death-time data shows P_S to be so sensitive a function of position in the abstract space that it is either substantially zero or unity. This is illustrated in Figure 2-5, which shows the sharp variation of P_S with particle diameter (D) and mass-infrared-emission coefficient (ε') for typical periapsis-ejecta-entry conditions. Analogously, there exists for every particle a critical speed (V_M), relative to Mars, above which sterilization is certain. Figure 2-6 shows V_M as a function of ε' and D at 1000 km. V_M is a function of altitude as well as particle properties and is zero above a critical altitude (h_m). The variation of h_m with ε' and D is shown in Figure 2-7. Based on this analysis a subset of all possible trajectories, called survival trajectories, can be identified. The fraction of all spores ejected on survival trajectories is P_I.

Preliminary Boeing spectrographic measurements of the emissivity of spore samples suspended in KBr pellets have produced ε' values of about 0.1 $cm^3/gm\text{-}\mu$. Figure 2-6 shows that for such ε' values the critical speed approaches periapsis orbital velocity. Further, as shown by Figure 2-7 all survival trajectories must necessarily originate within an altitude which is near the periapsis altitude of the presently-assumed highly-eccentric trajectory. The value P_I, the fraction of all spores ejected within the limiting range of velocities and directions, approximates the fraction of survival trajectories available while the spacecraft is within the altitude limits. This fraction is estimated to be less than 10^{-1}.

At present, the basic physical data needed to make the preceding analysis precise is lacking. A relatively small reduction in the value of the mass-emission coefficient (ε') would imply aerodynamic sterilization of all cold-propulsion exhaust and most cold-meteoroid ejecta particles. Therefore, there is an obvious need to refine the value of ε' and to determine the mass and speed distribution of the ejecta.

The second factor in the P_N term is P_{II}, which is the probability that an ejected spore which survives the aerodynamic environment will also survive exposure to the solar ultraviolet radiation during entry. Since all survival trajectories must originate near periapsis, as previously suggested for the given range of ε' values, a significant fraction of those trajectories will be occulted. Current studies of space survival on optically-thin particles indicate the solar ultraviolet 90 percent kill time near Mars is about 20 minutes; this is much less than the time for descent from periapsis of any particle in the 1 to 10 micron diameter range. However, the probability that a spore will simply not be exposed, due either to ejection and entry during occultation, or due to shielding by neighboring spores or other matter, appears to be relatively great. Therefore the probability P_{II} of surviving the solar UV during entry becomes essentially the non-exposure probability. P_{II} is presently estimated as 0.1 to 0.3. Additional analytical and experimental work must be performed before reliable estimates of these probabilities can be derived.

The third factor in P_3 is P_G, which is the probability that a spore which does survive ejection, entry and landing will propagate in the surface environment. Estimates by Sagan and Coleman on the probability that a given microorganism, landing on the surface of the planet would be able to multiply and contaminate a sizable fraction of the planet were 10^{-1} to 10^{-2}. Their estimates were based on evaluation of the expected "average Martian environmental conditions" considering primarily freeze-thaw cycle effects. More recent data from Evans of Goddard Spaceflight Center have indicated that ultraviolet radiation (2000 to 3000 Å) probably does reach the surface of Mars, thereby offering another hazard to microorganism survival on the planet's surface. This intense UV at the planet's surface may reduce the probability of growth by one order of magnitude.

From the inhospitable conditions of the Mars surface, as indicated by the Mariner IV data, the probability of spore germination and propagation may be assumed to be decreased by an additional order of magnitude. Therefore, a P_G of 10^{-3} appears to be a reasonable estimate at this time.

It is possible that the photo-decomposition of some materials under the influence of intense radiation as well as diffusion of spores from the internal structure may, over a long period, release spores into the surface environment. However, this presently appears highly improbable and was not considered in this analysis.

From the assumptions that P_N, the product of P_I and P_{II}, is less than 10^{-2} and $P_G = 10^{-3}$, the probability of contaminating Mars during the 50-year orbital-stay period is less than 2.8×10^{-5}. This approaches the allocated 1.0×10^{-5} allowable probability for planet contamination by meteoroid ejecta. If $P_G = 10^{-1}$, the probability of contamination is increased to a value less than 2.8×10^{-3} for the 50-year stay.

Considering the uncertainties in the assumptions used in this analysis, there is a possibility that the allocated probability could not be met without some spacecraft surface treatment. However, comparing the uncertainties of the calculations (the emissivity of the particles in particular) with the uncertainties of the Mars atmosphere and surface, it appears prudent at this time not to create "worst case" requirements which might force the system into a high-risk, low-reliability design or into a mission with a lower probability of success. Therefore, a system requirement for surface decontamination is not imposed at this time. The necessity for further study and experimentation to conform this position is apparent.

2.5.5 Propulsion and Reaction Control Subsystems

As part of Task A, analyses were performed to determine the probability of Mars contamination by ejecta from propulsion and reaction control subsystems. During Task B, these analyses have been refined and re-interpreted in the light of test results to evaluate planetary-quarantine requirements for the selected subsystems. The general equation for contamination probability given in Section 2.5.4 applies also in the present cases. Here, r represents the number of viable organisms that might appear in the exhaust (i.e., those in the propellant, or exposed interior surface, or imbedded in material that may spall), P_2 is the conditional probability that one of these individual organisms will be ejected alive, and P_3 is the conditional probability that a live ejected organism will contaminate Mars.

P_3 is again the product $P_I P_{II} P_G$. P_I is the probability that an individual organism will be captured and survive the entry aerodynamic environment; P_{II} is the conditional probability that an organism that survives the aerodynamic environment will survive solar ultraviolet radiation; and P_G is the probability of survival and propagation after reaching the surface of Mars.

The aerodynamic heating analysis leading to Figures 2-5 and 2-6 can be applied directly, leading to the result that P_I is the fraction of organisms ejected near periapsis and with velocity relative to Mars less than 1 to 2 kilometers per second in excess of orbital velocity. Moreover, P_{II} must be based on the set of organisms surviving aerodynamic heating and reduces to the probability that such an organism will be ejected and enter the atmosphere during occultation, or be shielded from ultraviolet by other organisms or inert material, or both.

Consideration of P_G is similar to that presented in Section 2.5.4. This is assumed to be in the range of 10^{-3} to 10^{-1}.

2.5.5.1 Reaction Control Subsystem

In the reaction control subsystem, r is substantially the number of organisms in the cold nitrogen propellant. No heating will occur, and, as shown in Table 2-8, cold nitrogen is not sporicidal. Therefore, P_2 is substantially unity.

Reaction-control-exhaust velocity relative to the spacecraft will be about 2000 fps and randomly directed. For the highly elliptic nominal orbit, the fraction of survival trajectories (Figures 2-5 and 2-6) reduces to the fraction of time that the spacecraft is sufficiently near periapsis. This leads to an estimated value of P_I of 0.1 for the nominal mission. Because the exhaust velocity is low, relative to the orbiting spacecraft, ejecta trajectory ranges will be long enough to preclude significant protection from ultraviolet due to occultation. Also, since the nitrogen must be filtered to reduce particulate contamination, effective shielding due to clumping is unlikely. Thus, P_{II} is estimated to be 10^{-3}. Further investigation of this phenomenon is required.

If $P_G = 10^{-3}$ is assumed, then $P_2P_3r = 10^{-5}\underline{r}$ which exceeds the probability allocation of 0.1×10^{-5} for all $r \geq 10$. Arguments similar to those in Section 2.5.4 may be presented regarding the uncertainties associated with the assumptions used in the analysis and with knowledge of the Martian environment. However, because it is relatively certain that viable organisms will be ejected, it appears prudent to plan for internal decontamination of the subsystem hardware and sterilization of the nitrogen gas by heat or filtration.

2.5.5.2 Midcourse and Orbit-Trim Engine

Previously published reports indicated that monomethyl and dimethyl hydrazine were lethal to Bacillus subtilis. An experimental program was conducted to extend these test results to other microbiological species and other propellants.

A series of the test propellants were inoculated with spores of Bacillus subtilis var. niger, spores of Aspergillus niger, and a soil suspension containing a representative population of soil microorganisms. Water controls were used. After inoculation, the propellants were stored at their respective standard storage temperatures and sampled at 24, 48, and 72 hours and at 7 days. The results are presented in Table 2-8 Test results indicate that hydrazine and aerozine are self-sterilizing.

Although hydrazine is sporicidal, an unsterile engine may still contain organisms that are not exposed to hydrazine. These microorganisms were not considered of significant contaminating potential to be included in the analysis. Midcourse firings should remove most of the contamination, thereby reducing the r value to a low figure. Furthermore, considering that these microorganisms will have to be wiped off by the hot exhaust gas, the probability of being ejected in a viable condition also appears to be very low. This remains to be verified by tests to be performed during Phase IB.

Table 2-8: SPORICIDAL-TEST SUMMARY

Sample Type	Results
Hydrazine	No survival after 24 hours; negative results after 20 days
Aerozine	No survival after 24 hours; negative results after 20 days
Nitrogen Tetroxide	Anaerobic survival
Liquid Nitrogen	No apparent reduction in viability after 14 days
Aluminized Polybutadiene	No apparent reduction in viability after 20 days
Water Control	Slight reduction in viability over 20-day period

The probability of contamination by orbit-trim-engine malfunction is equal to the probability of catastrophic failure (3.4×10^{-9}), which is well within the required constraint of 0.2×10^{-5}.

It appears that the midcourse and orbit-trim system will not require sterilization to satisfy the planetary-quarantine probability apportionment.

2.5.5.3 Orbit-Insertion Engine

During Task B, experimental investigations were performed of the sporicidal effects of aluminized polybutadiene. Test results showed no apparent reduction of inoculated microorganisms (B. subtilis spores and soil infusion) over a period of 20 days. The test also demonstrated that normally prepared propellant contains a natural flora.

Solid-propellant firing tests also were performed during Task B to determine whether microorganisms will survive the firing and be present in exhaust products. A series of 14 solid-propellant firings were conducted at the Boeing Tulalip facility. Seeded propellant was prepared by inoculating 36 grams of powdered aluminum with approximately 2×10^4 spores of Bacillus subtilis var. niger prior to mixing with polybutadiene. Firing tests used a 0.5-pound scale rocket motor. Prior to firing, the rocket motor, collection chamber, and associated pipes were steam-treated ($250^{\circ}F$ at 20 to 25 psig) for 30 minutes. The sterility of the 20-cubic foot collection chamber was checked and verified by microbiological analysis.

After sterilization, the solid-propellant grain was placed in the rocket motor and fired. A scrub-water system in the collection chamber was used to condense the exhaust products. Microbiological analyses of the scrub water and exhaust products were made. Results of the microbial analyses are given in Table 2-9.

The solid-propellant firing tests indicate a low probability that organisms in the propellant will be a source of contamination--in other words, P_2 approaches 0. The possibility of live ejecta from contaminated hardware remains to be established.

Table 2-9: RESULTS OF SOLID-PROPELLANT FIRING TEST

Number of Firings	Propellant	Igniter	Results
6	Seeded Aluminized Polybutadiene	Seeded Aluminized Polybutadiene	4 negative 2 positive*
2	Seeded Aluminized Polybutadiene	Naturally Contaminated Polyester Propellant	2 negative
4	Naturally Contaminated Aluminized Polybutadiene	Naturally Contaminated Aluminized Polybutadiene	4 negative
1	Seeded Aluminized Polybutadiene Surface Decontaminated with ETO Prior to Firing	Naturally Contaminated Polyester Propellant	1 negative
1	Seeded Aluminized Polybutadiene Surface Decontaminated with ETO Prior to Firing	Naturally Contaminaed Aluminized Polybutadiene	1 negative

* The positive results recorded appear to be caused by handling procedures during transporting of exhaust products from explosive safe area to the microbiology laboratory. Both samples showing positive growth were taken on the same day.

The probability of survival of microorganisms from the surfaces of nozzle internal surfaces appears low because dislodgement will be caused by the hot exhaust gases. The probability of existence of microorganisms in the ablative material of the nozzles also is low since the material is cured at temperatures and times lethal to the microorganisms. Furthermore, viable organisms ejected from the orbit-insertion engine will be traveling at high velocities relative to Mars, and the probability of a single spore surviving the entry heat pulse will be low. Consequently, this source of contamination is not considered significant in the analysis. The position remains to be confirmed by tests to be performed during Phase IB.

The arguments considering the probability of contamination by reaction-control exhausts are equally applicable to the thrust-vector-control exhausts since the excess freon will be dumped cold and the nitrogen is ejected cold. Therefore, to preclude Mars contamination, the TVC system should be decontaminated in the same manner as the reaction control system.

2.5.6 Conclusions

The following requirements are recommended to ensure compliance with the planetary-quarantine constraints: (1) sterilization of the orbit-insertion-engine TVC propellants and the interior of all associated TVC hardware, (2) sterilization of the interior surfaces of the reaction control subsystem and its associated propellant, (3) biasing of the trajectory aim point to prevent accidental spacecraft impact at encounter, and (4) selection of proper orbit parameters to ensure a nonorbit decay within 50 years.

During Phase IB, further studies are necessary to verify the conclusions and requirements resulting from the Task B effort:

1) Improve the determination of the emissivity value of spores and of organic and inorganic materials;
2) Determine the effects of solar radiation on various microorganisms--in particular, the photoreactivation and dormancy phenomena of spores after exposure to solar radiation;
3) Study the decomposition of material under Martian conditions and the sporicidal properties of decomposition products;
4) Examine the possibilities of spore erosion during entry--that is, conditions under which a spore loses atoms to an extent that would destroy it;

5) Extend studies of microorganism survival on propellants and in exhaust products;

6) Study the probable distribution of ejecta from both meteoroid impact and propulsion and reaction control subsystems by Monte Carlo techniques.

CONTENTS

		PAGE
3.0	SYSTEM-LEVEL FUNCTIONAL DESCRIPTION	3-1
3.1	Standard Trajectories	3-1
3.1.1	Transit Trajectories	3-1
3.1.2	Arrival Date Separation Considerations	3-11
3.1.3	Mars Orbit	3-16
3.1.4	Flight Capsule Entry Trajectory	3-39
3.2	Orbit/Entry Determination Uncertainties	3-41
3.2.1	Initial Mars Orbit Determination	3-41
3.2.2	Flight Capsule Entry and Impact Conditions	3-42
3.3	Voyager Flight Spacecraft Components Design Parameters	3-43
3.4	Voyager Equipment Elements and Documentation Identification	3-49
3.5	System Interface Definition	3-49
3.5.1	Interface of Spacecraft System and Capsule System	3-49
3.5.2	Interface of Spacecraft System and Launch Vehicle System	3-57
3.5.3	Interface of Spacecraft System and Tracking and Data System	3-57
3.5.4	Interface of Tracking and Data System and Mission Operations System	3-58
3.5.5	Interfaces of Spacecraft System and Mission Operations System	3-58
3.5.6	Interface of Capsule System and Mission Operations System	3-58
3.6	Telemetry Criteria	3-59
3.7	Telemetry Channel List	3-63
3.8	Guidance and Navigation Maneuver Error Allocations	3-63
3.8.1	Pointing—Angle Accuracy	3-65
3.8.2	$\triangle$V Magnitude Accuracy	3-65
3.9	Flight Sequence	3-65
3.9.1	Pre-Launch	3-68
3.9.2	Launch and Injection	3-68
3.9.3	Celestial Reference Acquisition	3-73
3.9.4	Interplanetary Cruise	3-76
3.9.5	Trajectory Correction Maneuvers	3-78
3.9.6	Flight Capsule Canister Separation	3-83
3.9.7	Mars Orbit Insertion	3-83
3.9.8	Planetary Vehicle Orbital Operations	3-85
3.9.9	Separation of Flight Spacecraft and Flight Capsule	3-88
3.9.10	Capsule Operations	3-88
3.9.11	Flight Spacecraft Orbital Operations	3-89
3.10	Preferred Spacecraft Layout and Configuration	3-89
3.10.1	Summary	3-89
3.10.2	General Description of Preferred Design	3-91
3.10.3	Selection of the Preferred Design	3-106
3.11	Selection of Preferred Spacecraft System	3-119
3.11.1	Summary	3-120
3.11.2	Evaluation Criteria	3-124
3.11.3	Selection Rationale	3-125
3.11.4	Reallocation of Spacecraft System Weights	3-128
3.11.5	System Concept Trade Studies	3-133
3.12	Planetary Quarantine	3-138
3.13	Voyager Flight-Equipment Cleanliness	3-140
3.14	Magnetics	3-141
3.14.1	Magnetic-Control Plan	3-141
3.15	Space-Radiation Effects on Voyager System	3-145
3.15.1	Semiconductor-Device Response	3-146
3.15.2	Voyager-System Materials	3-146
3.16	Data System Description	3-147
3.17	System Reliability Summary	3-155
3.17.1	Reliability Requirements	3-159
3.17.2	Preferred System Evaluation	3-162
3.17.3	Subsystem Evaluation Summary	3-177

3.0 SYSTEM-LEVEL FUNCTIONAL DESCRIPTION

This section contains the system-level functional description of the preferred spacecraft design and its relationship and interfaces with other mission elements. A set of trajectories, limited by mission constraints, is described. A Spacecraft System is defined in broad enough terms to allow identification of system interfaces. A top-level flight sequence is developed from which the preferred spacecraft design is evolved--a result of many system and subsystem trade studies. Special system-level requirements and constraints, such as planetary quarantine, cleanliness, magnetics, radiation, and data system, are presented. A reliability analysis and assessment of the design against the reliability allocations concludes the section.

3.1 STANDARD TRAJECTORIES

This section describes the range of trajectories, considerations required to select meaningful trajectories, and the most useful parameters of typical trajectories.

3.1.1 Transit Trajectories

3.1.1.1 Engineering Design Considerations

Constraints on Launch and Arrival Dates--The Type I trajectories available for use in 1971 are those that have launch and arrival dates within the boundaries depicted on the base design chart, Figure 3.1-1 (see Reference 1, Figures 90-95). Three of the boundaries are established

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Reference 1: JPL EPD No. 281, "Trajectory Selection Considerations for Voyager Missions to Mars during the 1971-1977 Time Period," C. E. Kohlhase and W. E. Bollman, September 15, 1965.

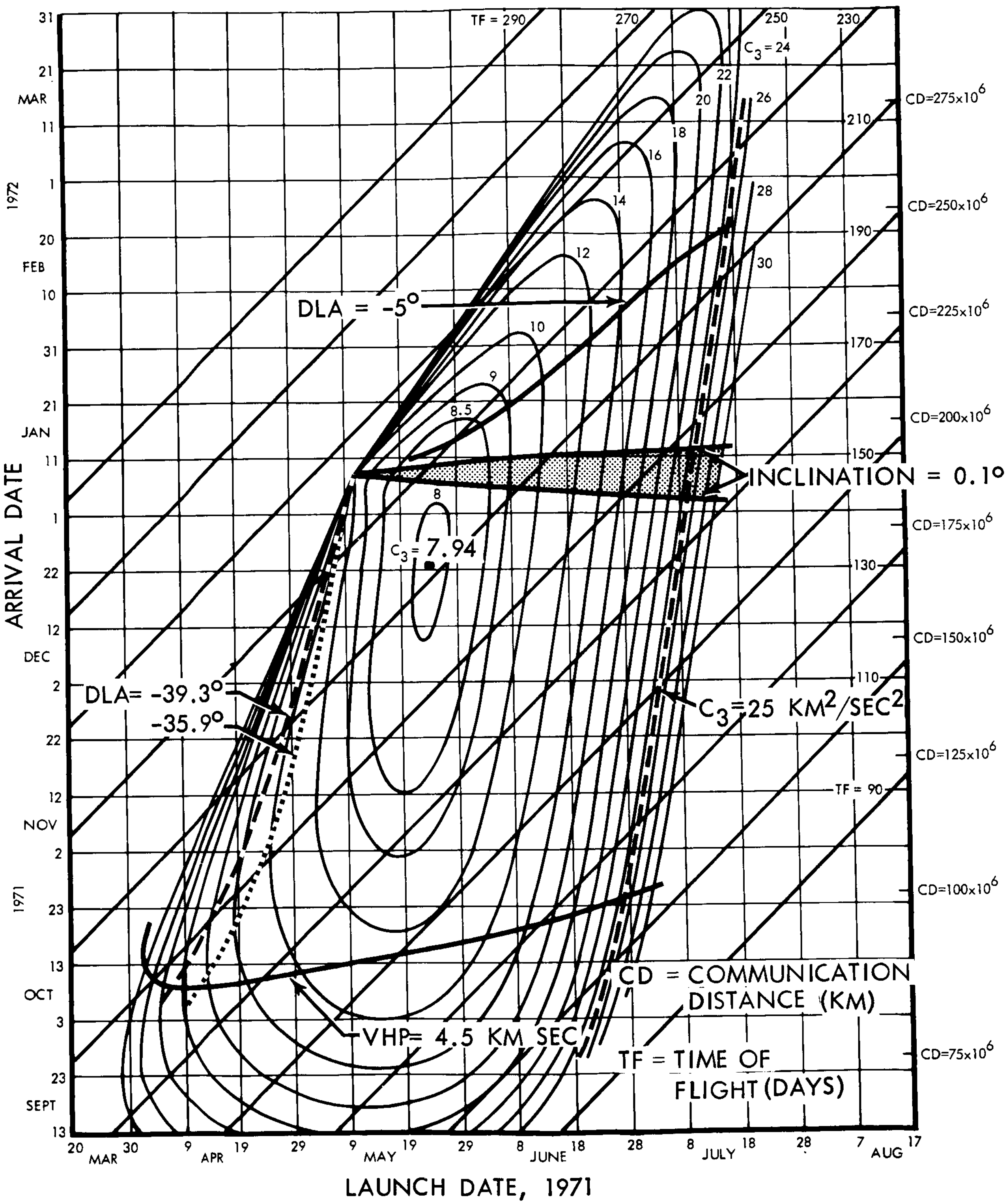


Figure 3.1-1: Trajectory Design Chart, 1971 Type I

directly by the specified constraints: maximum vis-viva energy, $C_3 = 25\ km^2/sec^2$; minimum absolute value of the declination of the departure geocentric asymptote, $|DLA| = 5$ degrees; and maximum hyperbolic excess speed at Mars, VHP = 4.5 kilometers per second. The DLA boundaries on early launches result directly from the following requirements:

1) Latitude of the launch site (Complex 39 at Kennedy Space Center) = 28.65 degrees;
2) Allowable range of launch azimuths = 60 to 115 degrees;
3) Two-hour minimum daily firing window.

For northeast launches, the boundary is determined by $|DLA| < 39.3$ degrees; for southeast launches, by $|DLA| < 35.9$ degrees. The negative value of DLA applies in both cases because of the requirements of the Type I transit trajectories. Because northeast launches produce shorter coast times in the parking orbit, the corresponding limit (DLA = -39.3 degrees) will be generally used. The limit applicable to southeast launches, DLA = -35.9 degrees, is shown for reference. (The early launch boundary may be modified slightly by a limitation on the velocity increment available for biasing the arrival date. This is discussed briefly in Section 3.1.2.)

A further requirement for the Voyager mission is the minimum launch period of 45 days. This is easily satisfied because the range of launch dates shown on Figure 3.1-1 between the limits DLA = -39.3 degrees and $C_3 = 25\ km^2/sec^2$ varies from 61 to 93 days. Therefore, there is considerable flexibility in the choice of launch and arrival dates.

The region of available trajectories is divided by the exclusion of inclinations of the heliocentric transfer orbit that lie between 0.1 and 0 degrees. Operation in the region of arrival dates later than January 11, 1972, is considered both unnecessary and undesirable. It would require generally longer trip times and higher trajectory sensitivities, and the region of earlier arrival dates has a sufficient range of launch dates.

The overall effects of the changes in the constraints from those of Task A have been to increase the range of launch dates by about 11 days and to delay earliest arrivals about 6 days.

Launch Window--Launch window considerations for Task A constraints and requirements were discussed on Page 3-17 of Reference 2. The results were illustrated in Figure 3.1-6 of Reference 2.

For Task B, increases were made in the allowable ranges of launch azimuth, DLA and C_3, as discussed above. In addition, a new maximum parking-orbit coast time of 90 minutes was specified. These changes produce several effects. The change of the DLA limit from -33 degrees to -39.3 degrees provides a total DLA range from about -18 degrees to -39.3 degrees. The launch window available at a given value of DLA

Reference 2: Boeing Document D2-82709-1, Part I, "Voyager Spacecraft System Final Technical Report," Vol. A, Preferred Design for Flight Spacecraft and Hardware Systems.

has been increased by widening the allowable launch azimuth sector (this increase is somewhat more for northerly launches than for southerly launches). The increase in the available range of C_3 values widens the range of required parking-orbit coast times. On the other hand, the available parking-orbit coast time essentially removes the coast time limitation and provides two launch windows in each 24-hour period at most declinations. With one launch window is associated a long coast time, with the other a short coast time. Other factors now become important. Short coast times provide maximum injection accuracy, maximum system reliability, and minimum propellant boiloff. The use of both launch windows would require the use of two widely separated injection locations with attendant duplication of tracking capability. These considerations favor the use of only the short parking-orbit coast time and the associated northeast launch azimuths.

Communication Between Planetary Vehicles and Earth--Each Planetary Vehcile will communicate with the Earth-based DSN via the low-gain omnidirectional antenna until it reaches a distance of approximately 45 million km from Earth. Then, at as late a time as warranted by the performance of the telecommunication subsystem, the change will be made to the high-gain directional antenna. Figure 3.1-2 presents the envelope of distances and nominal times of operation for the low-gain and high-gain antennas on available transit trajectories in 1971. In Figure 3.1-3, the pointing requirements for the antennas are illustrated by the envelope of Earth cone and clock angles for points along all available transit trajectories from 10 days to encounter. The cone-

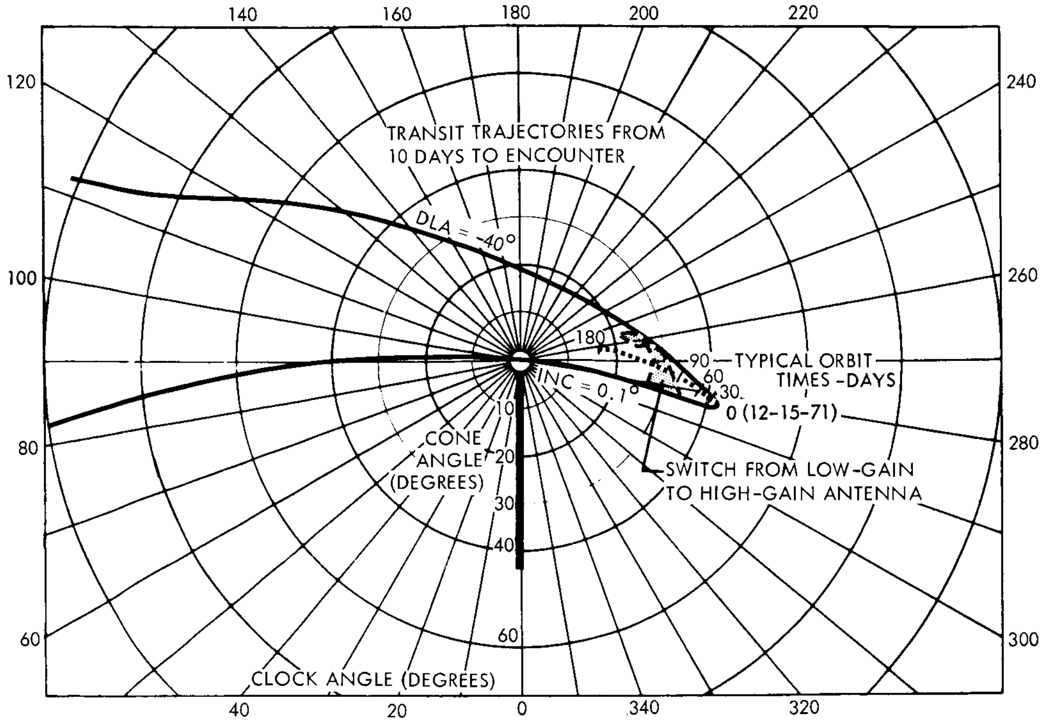


Figure 3.1-2: Communication Distance Envelope

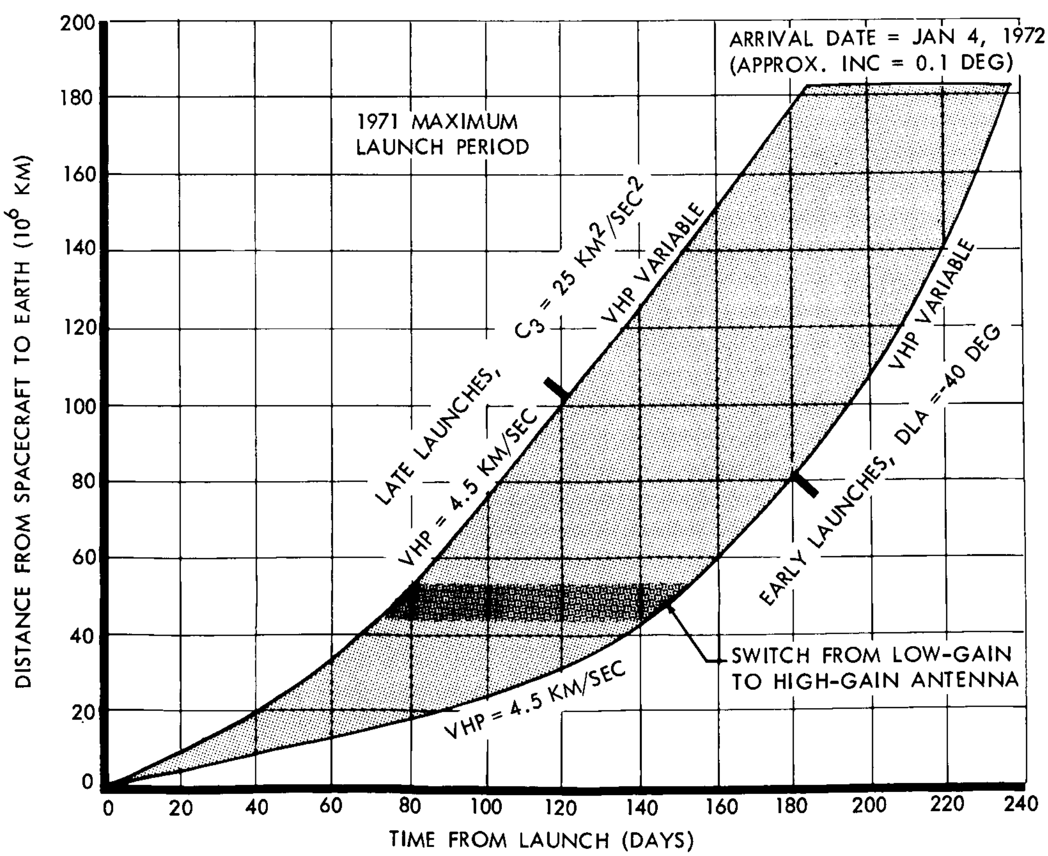


Figure 3.1-3: Envelope Of Earth Cone And Clock Angles

clock angle region for switching from the low-gain antenna to the high-gain antenna has been mapped from the distance versus time chart. It is of particular interest that the required operational envelope of the high-gain antenna comprises a relatively small portion of the total envelope. The data for Figures 3.1-2 and 3.1-3 were computed from 67 trajectories throughout the constrained region by a special-purpose conic program. A limited check with values produced by a six-body integrating trajectory program indicates an agreement generally within 0.5 degree on Figure 3.1-3.

Approach-Aiming Parameters for Transit Trajectory Design--The approach-aiming coordinates, defined in Figure 3.1-4, describe the position of Sun and Earth with respect to the approach-aiming-coordinate system. This system is based upon the direction of the approach asymptote, given by $\overline{S}$, and the axis $\overline{T}$, parallel to the ecliptic plane. The Sun angular coordinates ZAP and ETS are plotted in Figure 3.1-5 for the available 1971 launch and arrival dates. The Earth coordinates ZAE and ETE are similarly presented in Figure 3.1-6. The data for all these figures were obtained from Reference 1.

Some of the requirements and objectives of the areocentric phase of the mission can be transformed into requirements for the transit trajectory by means of the approach-aiming coordinates. From Figure 3.1-4 it is apparent that Sun and Earth maximum occultations occur whenever aiming points are chosen in the $\overline{R}$-$\overline{T}$ plane along the lines determined by the angles ETS and ETE, respectively. The ETS values of Figure 3.1-5 indicate that no Sun occultation exists on approaches to direct orbits about Mars. The ETE distribution in Figure 3.1-6, however, shows Earth

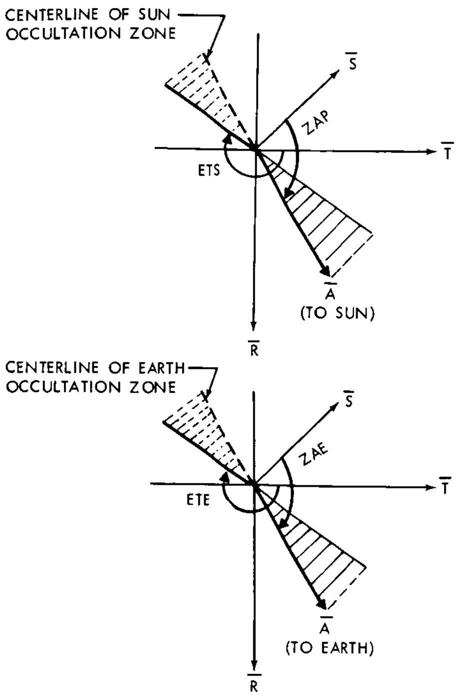


Figure 3.1-4: Definition of Approach Aiming Coordinate System

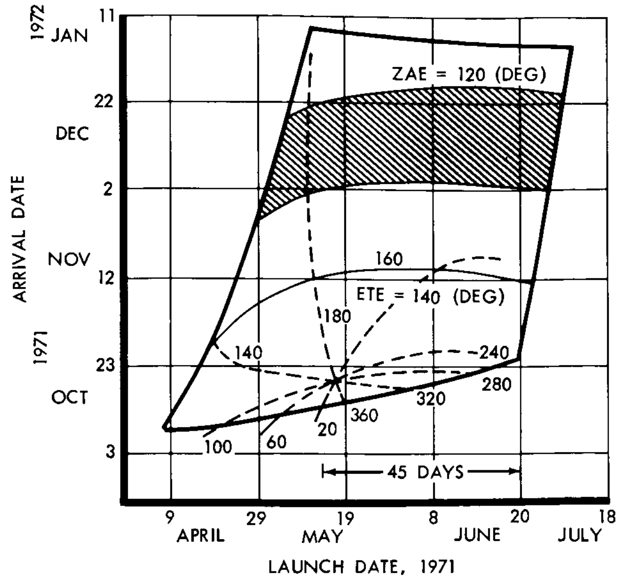


Figure 3.1-6: Orientation Of Mars-Earth Line In The Approach Aiming System

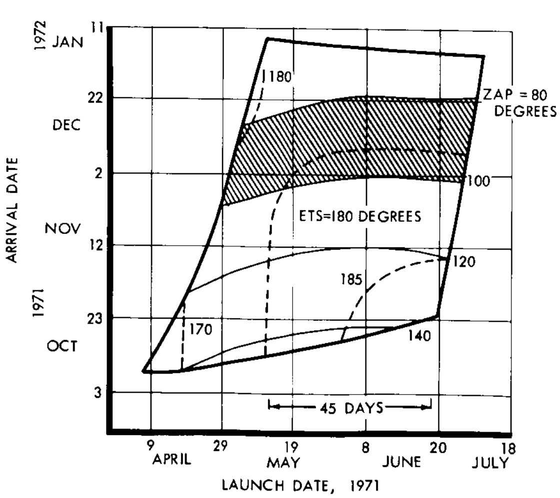


Figure 3.1-5: Orientation Of Mars-Sun Line In The Approach Aiming System

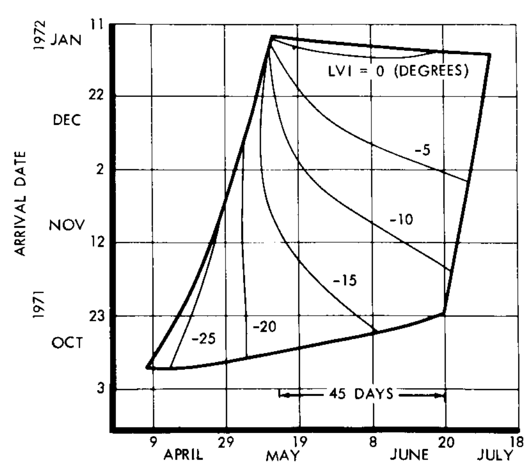


Figure 3.1-7: Latitude Of Vertical Impact Point On Mars, LVI

occultations occur on approaches to direct orbits for transit trajectories with early arrival dates launched from late April to early June. Greater detail for specific approach trajectories must be obtained from occultation charts like Figure 3.1-21 of Reference 2, which also shows the Canopus occultation zone. The lack of difficulty with Earth and Sun occultations on the approach trajectories implies that occultations occurring during the orbital phase will play a more important part in the trajectory design. These are discussed in a later section.

Angle ZAP indicates whether the approach to the planet is from the sunny side (ZAP greater than 90 degrees) or dark side (ZAP less than 90 degrees). It also gives an indication of the position of the approach-hyperbola periapsis relative to the Mars-Sun line since the areocentric angle from S-vector to periapsis varies from about 55 to 75 degrees. Together with this angle, ZAP shows the amount of in-plane rotation necessary to place the orbit periapsis at a desirable illumination.

The angle ZAE describes the orientation of the approach asymptote relative to the Mars-Earth line. Since the Planetary Vehicle is Sun-oriented, differences in the variations of ZAP and ZAE over the launch period indicate varying Earth-pointing requirements for the antenna. A comparison of Figures 3.1-5 and 3.1-6 shows that major differences occur only for trajectories with early arrival dates.

Scientific experiment requirements and the more detailed consideration of the approach phase in Section 2.4 and the orbital phase in Section 3.1.3 are of primary importance in the selection of the set of launch and arrival dates. Also, when possible, it is desirable to simplify the mission requirements by achieving the minimum variation in arrival geometry with respect to Earth and Sun. The launch and arrival dates which best achieve this are shown as shaded areas in Figures 3.1-5 and 3.1-6.

3.1.1.2 Scientific Considerations

The objectives of probable orbiter scientific experiments generate three main requirements for the areocentric phase of the mission:

1) Latitude of the subperiapsis region in the range of plus 10 to minus 40 degrees;
2) Relatively low illumination angles in the subperiapsis region;
3) Observation of the wave of darkening phenomenon, if possible.

The approach parameter LVI, the latitude of the vertical impact point on Mars for the approach trajectory, defines the minimum inclination available for the approach hyperbola. When near-equatorial orbits are wanted, the small values of LVI in the upper right region of Figure 3.1-7 should be used to avoid excessive orbit-trim requirements.

Low illumination angles are obtained with lowest orbit-insertion ΔV requirements by placing the periapsis of the approach hyperbola near the terminator. When angle ZAP is 90 degrees more than the angle between the S-vector and the hypobola-periapsis radius, the periapsis is in the terminator plane. Since this angle typically will vary between 55 and 75 degrees, a satisfactory illumination angle of 15 degrees can be obtained with ZAP values between 130 and 150 degrees. The larger ZAP values correspond to higher VHP values and/or higher periapsis altitudes. This requirement leads to the use of early arrival dates in the region toward the bottom of Figure 3.1-5 and produces a conflict with the trends discussed in the previous paragraphs. The Planetary Vehicle, however, has the prescribed capability to rotate the line of apsides of the elliptical orbit in the orbit plane $\pm$ 90 degrees from the hyperbola periapsis. This reduces the conflict to a varying increase in orbit insertion ΔV. In general, the smaller ZAP angles will require larger expenditures of orbit insertion ΔV to produce the desired low illumination angles.

The observation of the wave of darkening phenomenon was discussed in Task A report, Reference 2, Section 3.1.2. It was shown that early arrival dates may be necessary to achieve best results.

3.1.2 Arrival Date Separation Considerations

The Planetary Vehicle arrival dates at Mars are required to be separated by at least 10 days. Each vehicle is provided with a total velocity increment (ΔV) capability of 200 meters per second to perform all of the necessary interplanetary corrections and biasing. A ΔV allotment of about 50 meters per second each should be adequate for all midcourse corrections and aiming-point biasing under the adaptive guidance policy. This leaves about 150 meters per second for each vehicle or a conservative total of about 300 meters per second for arrival-date biasing.

The available ΔV is used most economically by combining the maneuvers required to perform the first midcourse correction, aiming-point biasing, and the arrival-date biasing. This procedure also reduces the total number of maneuvers, involving loss and reacquisition of celestial references, reorientation, and thrusting. Performing the maneuver together with the first midcourse correction will permit monitoring of engineering data via the low-gain antenna in the event of a malfunction.

Figures 3.1-8 and 3.1-9 present a few representative curves that illustrate the ΔV requirements for the arrival-date biasing maneuver, performed 5 days after launch. Figure 3.1-8 illustrates the ΔV required for one Planetary Vehicle to achieve the entire 10-day bias. Figure 3.1-9 illustrates the ΔV requirements for each of the two vehicles to perform a 5-day bias. A dual symmetry exhibited by the biasing ΔV requirements is illustrated schematically in the sketch on Figure 3.1-9.

The data of the figures were calculated separately for each bias by a newly developed special-purpose conic program. The ΔV values for the symmetric cases agreed, in general, within 3 meters per second. A check of the conic data with two widely separated cases computed with a six-body integrating-trajectory program gave good agreement.

The approximate maximum ΔV capability indicated in Figure 3.1-8 shows that one vehicle is capable of performing the entire biasing maneuver for many trajectories. This suggests that a single-vehicle mode should be used in some instances. Such a procedure would have the advantage of

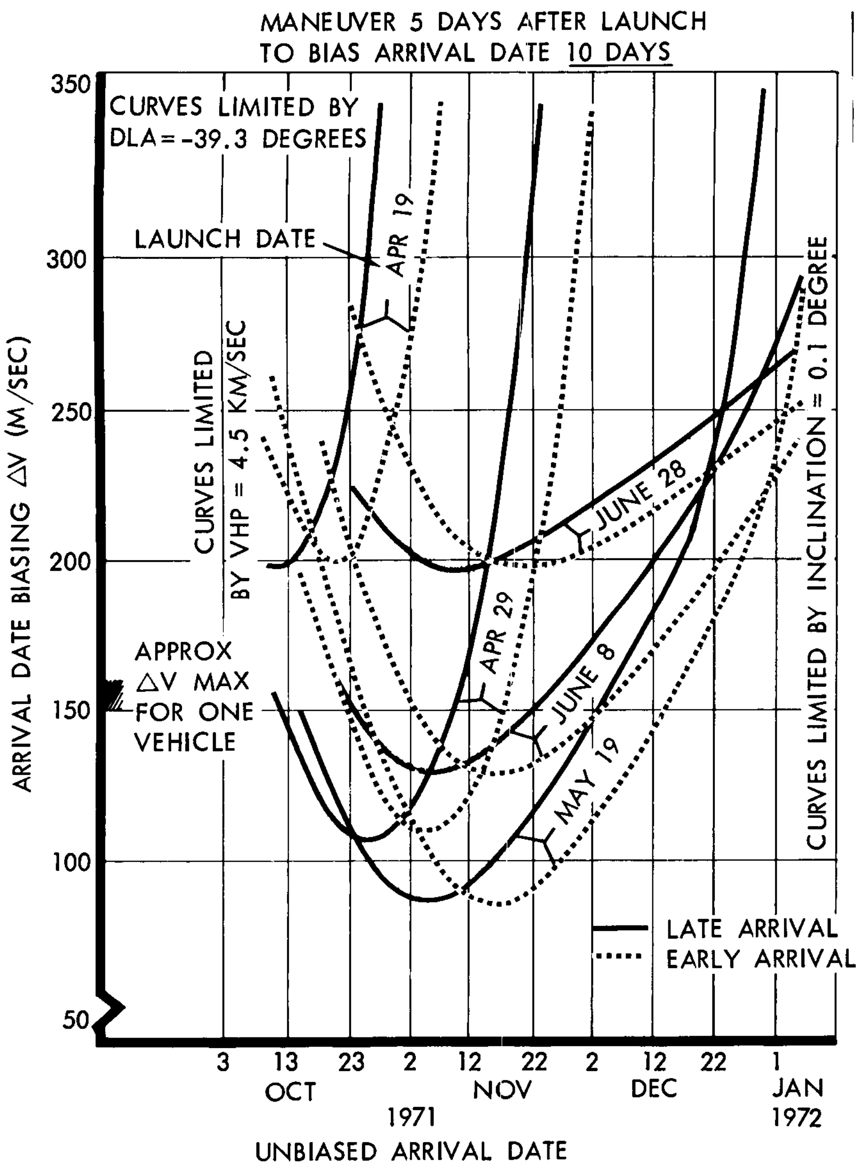
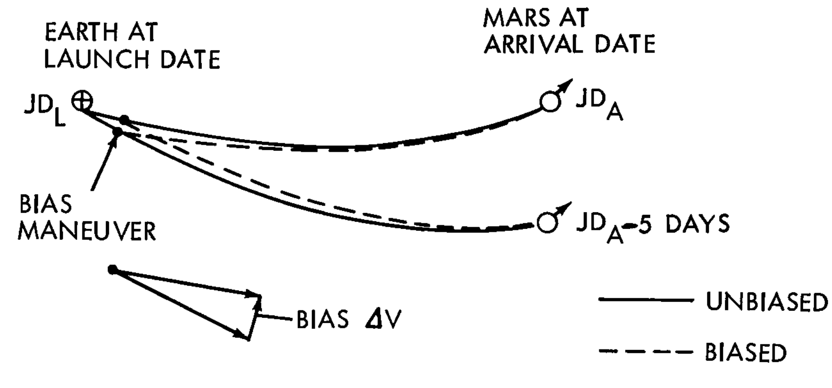


Figure 3.1-8: Arrival Date Biasing Requirements For One Vehicle Maneuver



Biasing Symmetry

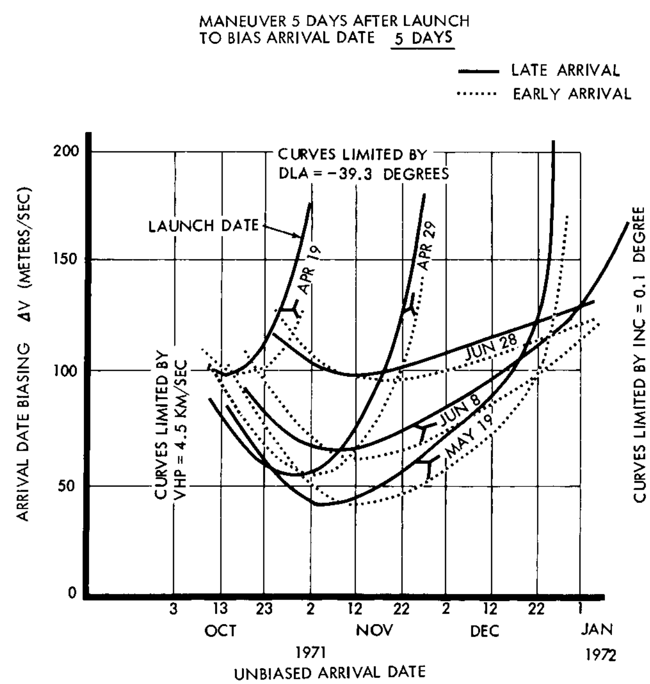


Figure 3.1-9: Arrival Date Biasing Requirements for Two-Vehicle Maneuver

requiring only one vehicle to make the relatively large trajectory adjustment. Thus, the probability of subsequent midcourse corrections would be increased for only that vehicle.

The capability of a single Planetary Vehicle to perform a 10-day biasing maneuver has been mapped into ΔV contours on the constrained-trajectory design chart in Figure 3.1-10. For all trajectories with launch and arrival dates within these contours, the indicated ΔV capability is sufficient to achieve the 10-day arrival date bias. Figure 3.1-10 shows that one Planetary Vehicle can adequately bias the arrival dates of approximately 35 percent of the trajectories considered. The 10-day biasing maneuvers can be performed for most of the trajectories with launch dates between April 24 and June 15 and arrival dates before December 16.

The two-vehicle biasing-maneuver capability is illustrated in Figure 3.1-11. It exhibits characteristics quite similar to those of Figure 3.1-10, except that the required 5-day biasing maneuvers are shown to be within the capability of the two vehicles for most of the constrained launch-arrival-date region. At the earliest launch dates, the biasing-performance envelope falls slightly inside of the previously established DLA boundary. It is interesting that the biasing ΔV varies linearly with the arrival-date bias requirement. For example, the 5-day bias contour for ΔV = 50 meters per second almost coincides with the ΔV = 100 meters per second contour for a 10-day bias.

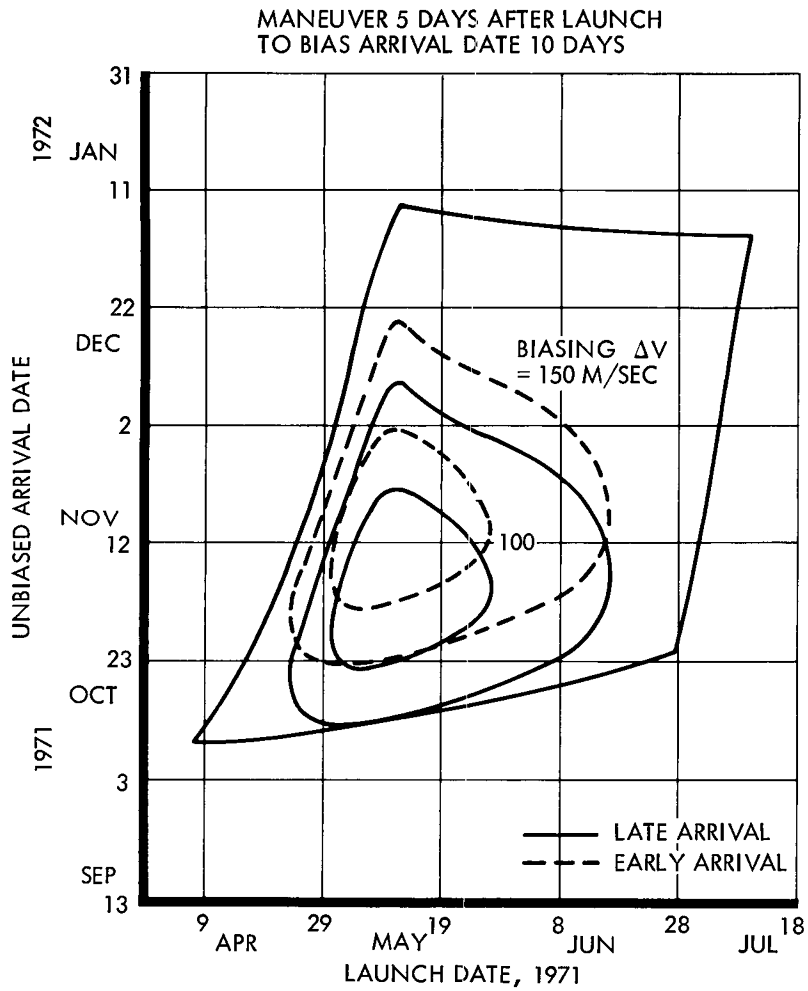


Figure 3.1-10: Arrival Date Biasing Capability for One-Vehicle Maneuver

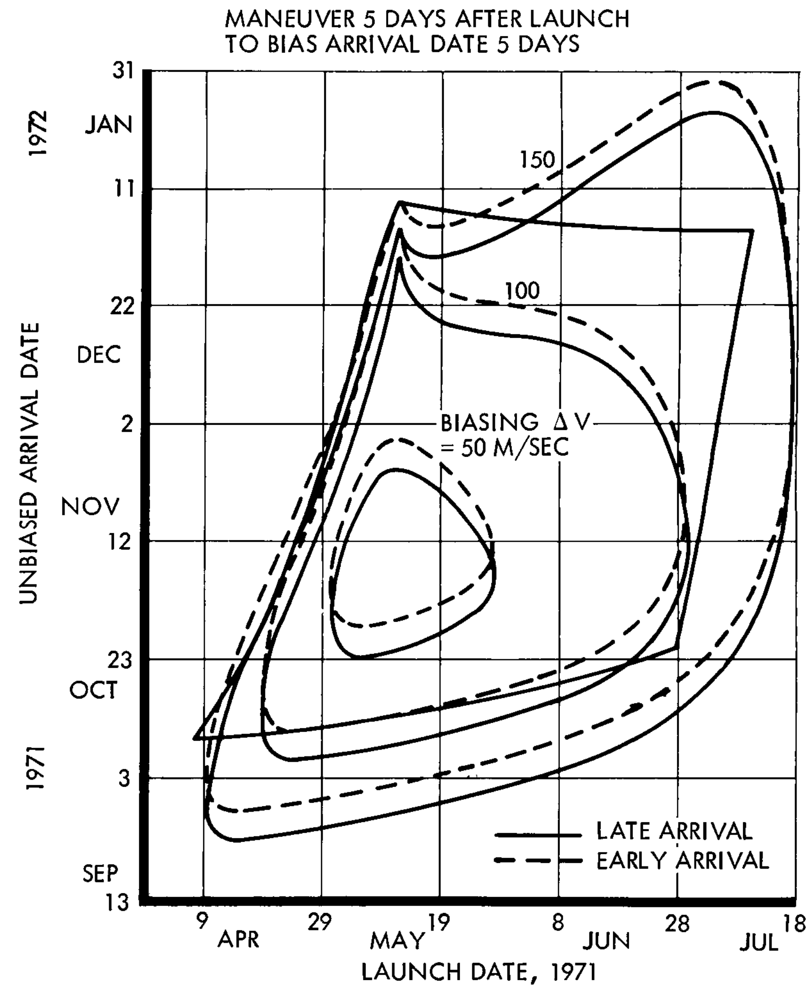


Figure 3.1-11: Arrival Date Biasing Capability of Each Vehicle in Two-Vehicle Maneuver

3.1.3 Mars Orbit

3.1.3.1 Approach Geometry

Considering the transit trajectories within the study constraints, a wide variation exists in the direction of the approach asymptote. The right ascension and declination of the approach asymptote (S-vector) will each vary over approximately a 30-degree range. This S-vector direction determines the lowest possible orbit inclination about Mars, and, to a large degree, the requirements to obtain a particular periapsis position for the orbit about Mars. Typical view angles to Mars are described in the cone-clock-angle plot of Figure 3.1-12. The range of cone and clock angles of Mars at approach along the S-vector is shown by the shaded region outlined with the dashed curve. Three example approach trajectories have been shown for a particular S-vector corresponding to a launch on May 1, 1971, and an arrival on November 1, 1971. These approach trajectories have orbit inclinations (with respect to Mars equator) of 20, 40, and 60 degrees. The view angles to Earth and Canopus are also shown during the approach phase.

3.1.3.2 Insertion

The location of insertion into orbit about Mars is determined by the particular orbit desired. Various techniques to perform the insertion maneuver are available to obtain a desired orbit size, location of the line of apsides, and inclination to the Mars equator. The advantages and range of applicability of the more promising techniques are discussed below.

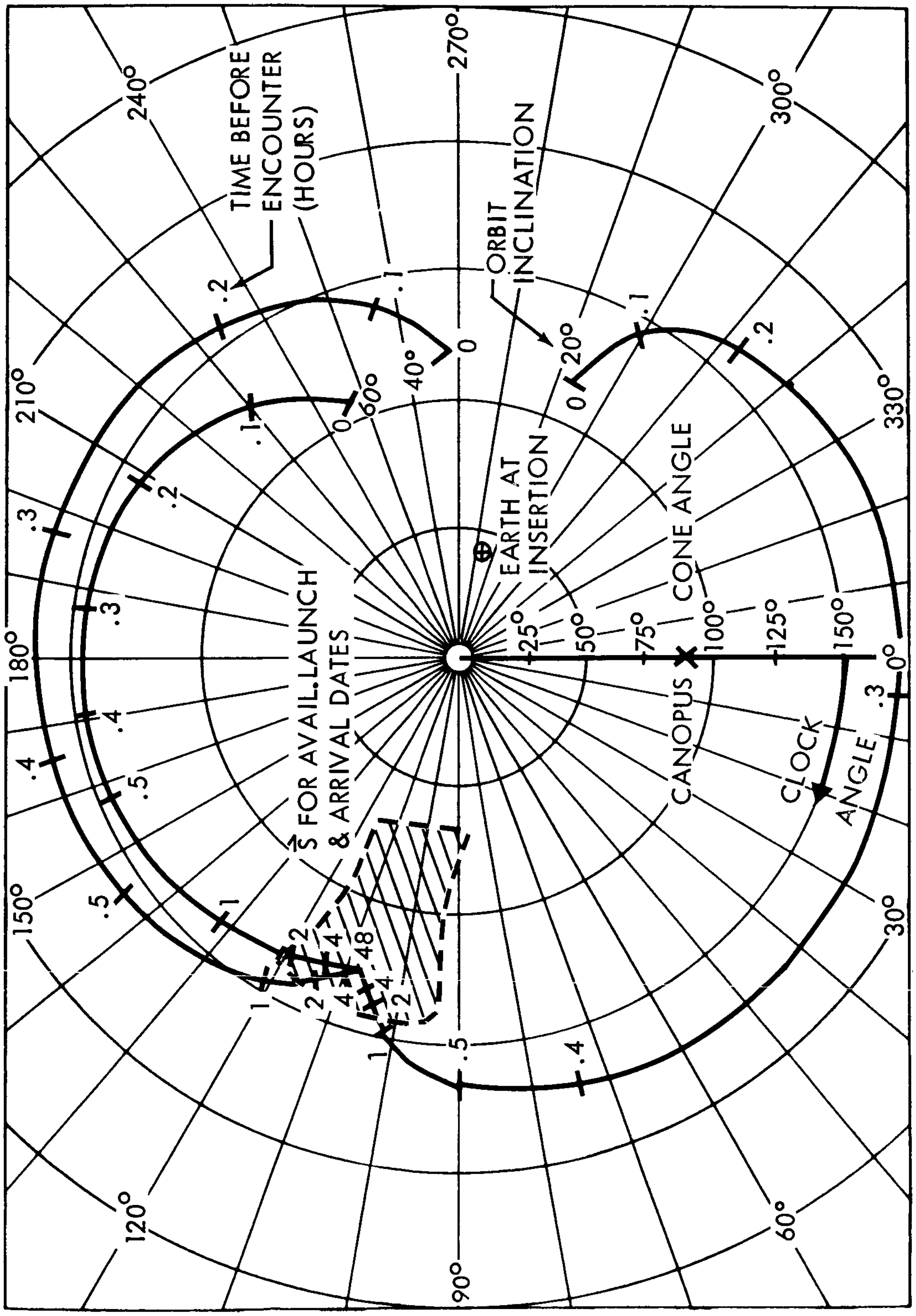


Figure 3.1-12: View Angles at Approach to Mars

Insertion Techniques--The most efficient technique involves operation of the insertion motor at periapsis of the approach hyperbola, resulting in the orbit periapsis at the same point. Figure 3.1-13 shows the orbits that are available with insertion at periapsis. The lower limit on the curve indicates the minimum size orbit, from a lifetime consideration, that is available using the realistic atmosphere presented in Reference 1. The available orbits that can be obtained are shown for the range of VHP of interest and two propulsion-system capabilities. Those orbits which fall to the right of a given VHP curve may be obtained from approach at the VHP. For scientific experiments, however, other orbit orientations are preferable. Therefore, another periapsis point must be provided for the satellite orbit. Figure 3.1-14 shows the minimum ΔV requirements to insert into an example orbit with varying positions of the line of apsides. This insertion is accomplished as shown in the sketch attached to the figure. The nominal aiming point is selected to provide a hyperbola periapsis at an altitude above the periapsis of the Mars satellite orbit. At the proper point along the hyperbola (either before or after hyperbola periapsis), the insertion-velocity change is applied. This impulse generates a new velocity vector that corresponds to the proper point on the desired satellite orbit. The hyperbola aiming point and the magnitude of the velocity change and its direction are selected to provide a satellite orbit with the desired size, shape (that is, periapsis altitude and orbit period), and orientation. The example orbit of Figure 3.1-14 has a periapsis altitude of 1000 km, an orbit period of 13.8 hours, and is entered from various approach speeds (VHP). The ΔV requirements are

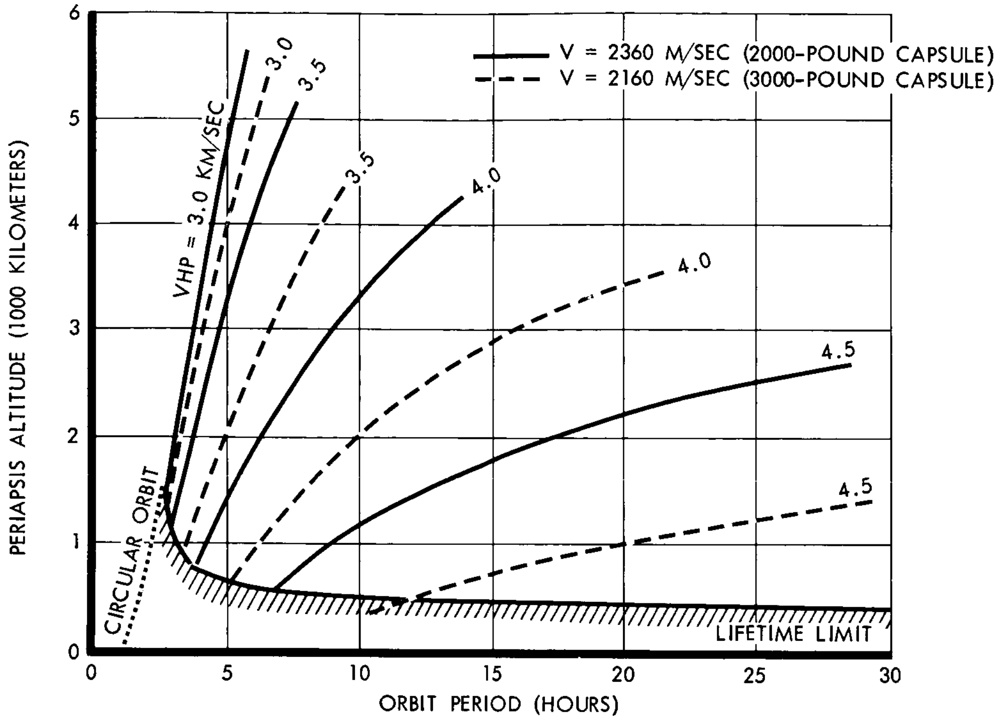


Figure 3.1-13: Available Orbits For Insertion at Periapsis

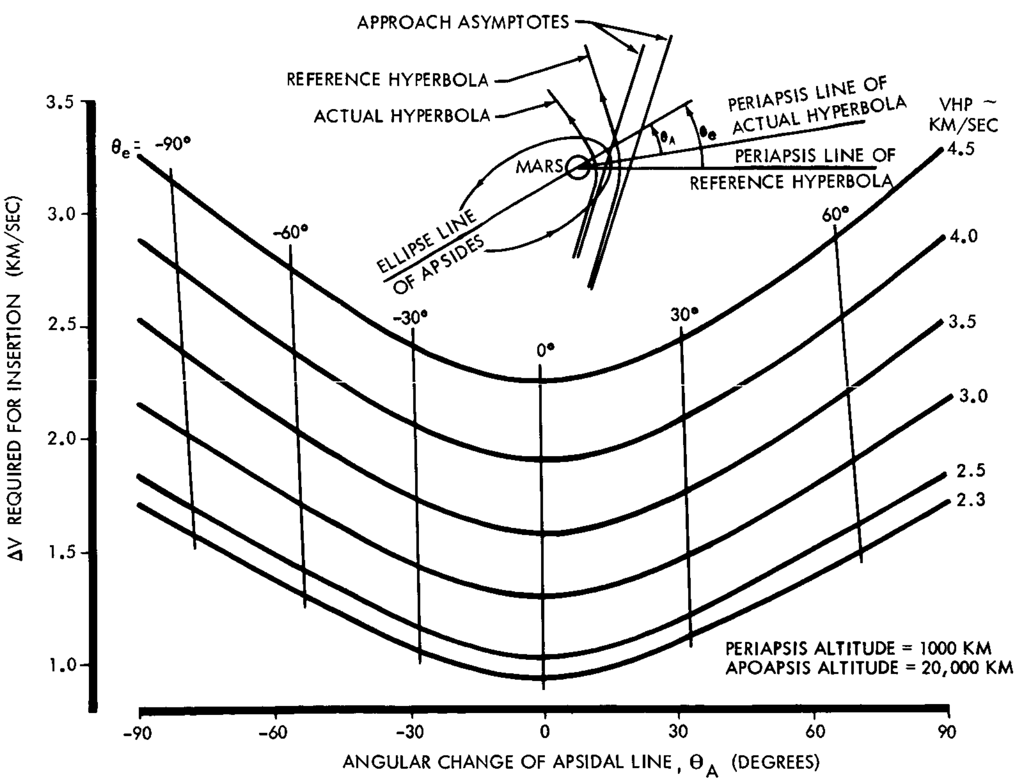


Figure 3.1-14: Velocity Requirements for Orbit Insertion

increased for the higher values of VHP and for the larger rotation angles.

The typical insertion geometry is represented in the cone-clock angle plot of Figure 3.1-15. This curve has the same S-vector as shown on Figure 3.1-12. A nominal inclination of 40 degrees has been selected for this example. The center of Mars as it appears from the spacecraft during the 2 days prior to insertion is shown by the solid curve. The two dashed curves represent the near and far limbs of Mars. The total angle subtended at the Planetary Vehicle between these two limbs at closest approach is approximately 100 degrees. The required orientation of the insertion engine to provide the necessary ΔV is shown for various requirements of rotation of the line of apsides. The cone-clock angle plot also shows the position of Earth and Canopus. Note that the trajectory approaches in a direction that will not interfere with the Canopus tracker.

Some gain in knowledge of the spacecraft position can be obtained with insertion after periapsis because trajectory determination can be continued slightly longer and Mars will cause a more significant change in spacecraft velocity. This will provide better navigation data from the DSN. This also provides the ability to define more accurately the timing and magnitude of the velocity impulse. At present, it is not certain that a significant increase of satellite-orbit accuracy results from insertion after periapsis.

Variable-Impulse Insertion--Figure 3.1-14 showed the amount of apsidal rotation that can be obtained for an example orbit size. This apsidal

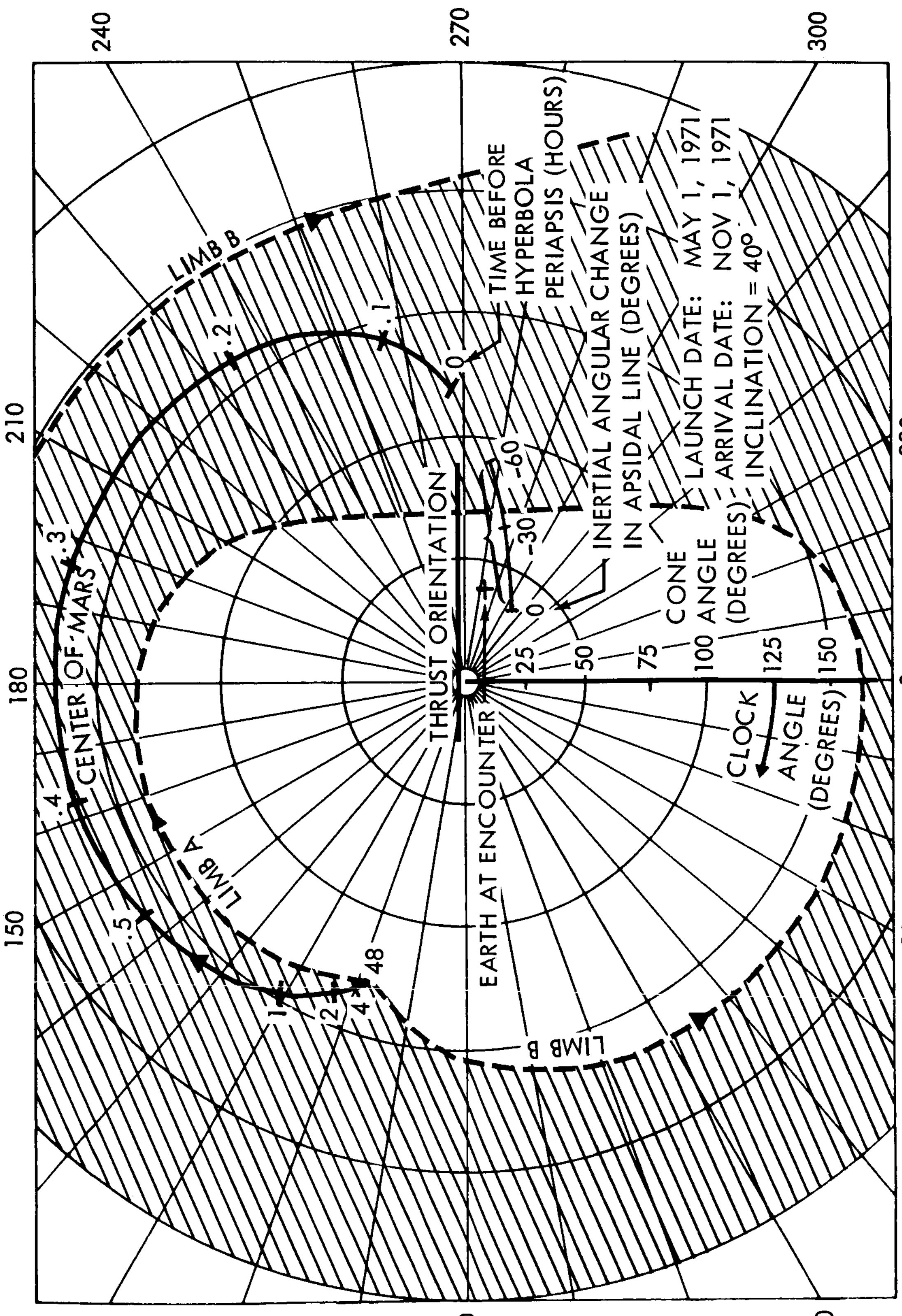


Figure 3.1-15: Insertion Geometry

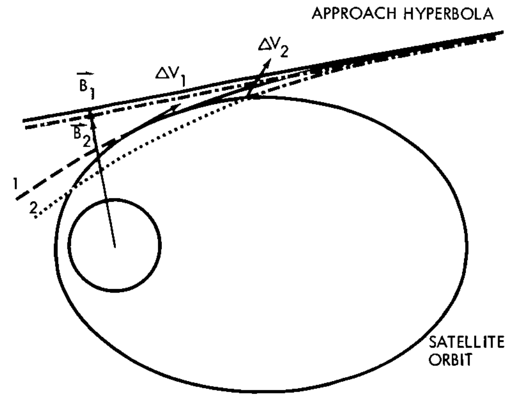
rotation can be measured in two ways, as shown on the sketch. The data presented in Figure 3.1-14 has been referenced to an arbitrary inertial direction. This inertial system has its reference axis located at periapsis of a hyperbola with VHP of 3.0 km/sec and periapsis altitude of 1000 km. It is computationally more efficient to reference the angle change in the orbit line of apsides to the periapsis of the approach hyperbola for the particular insertion considered. However, this method does not provide a complete and convenient inertial reference for the periapsis because the reference line changes its orientation to the Sun as the B-vector size and VHP change. For this latter case, a ± 90 degree variation in orbit is less than ± 90 degrees when measured in inertial space. Because the inertial direction is believed to be more meaningful for mission planning, the subsequent orbit rotation charts will be referenced to the arbitrary inertial direction listed above. This is the first step in applying data of the type shown in Section II.D.4 of Reference 1. The most significant advantages of variable-impulse insertion are the capability to change the insertion velocity impulse, to allow optimum insertion into all orbits with varying VHP, and to account for known dispersions in the B-vector.

Fixed-Impulse Insertion--If a fixed-impulse motor is used in the spacecraft design, the propulsion subsystem is sized to accommodate the largest ΔV that is required through consideration of mission requirements on apsidal rotation and VHP during the launch period. The excess propellant must be purposely wasted when the transit trajectory and orbit are such that an optimum insertion maneuver requires less ΔV than the motor provides. One method available for using up this excess propellant is shown in Figure 3.1-16. For a given approach hyperbola. there is a range of aiming points (B-vectors) to establish the same satellite orbit, each with a different ΔV requirement. It then follows that if B is varied as VHP is varied over some range, the ΔV requirement for establishing the orbit can be maintained at a constant value. This method can be used to provide for a constant orbit orientation and size even with varying launch and arrival dates. At the proper time from periapsis, the insertion ΔV is applied. The direction of the fixed impulse is selected to provide the proper orbit. This pointing problem is no more complicated than that for optimum variable impulse insertion. The option exists to perform the insertion maneuver on either side of the periapsis for the approach hyperbola. This technique can be used so long as the hyperbola periapsis can be maintained above a minimum value selected on the basis of planetary impact considerations. Figure 3.1-17 shows the ΔV requirements for establishing a typical orbit at various apsidal locations.

Figure 3.1-18 is similar, except the orbit shown now has a low periapsis altitude and a high apoapsis altitude. This is the type of orbit that is most difficult to accommodate with a fixed-impulse design. In the event that the periapsis of such an orbit is too low to allow usage of the full-orbit-insertion engine impulse, an in-orbit vernier capability such as orbit trim can be used to advantage. For such a case, an intermediate orbit is entered with a higher periapsis.

3.1.3.3 Satellite Orbit about Mars

Orbit Selection Considerations--The orbit parameters can be divided into two categories: those items that determine the size and shape of



$\vec{B}_1$ = AIMING POINT FOR MINIMUM ΔV_1 INSERTION WITH SPECIFIC APSIDAL LINE

$\vec{B}_2$ = AIMING POINT FOR SAME SATELLITE ORBIT WITH SATELLITE ORBIT WITH $\Delta V_2 > \Delta V_1$

Figure 3.1-16: Insertion Maneuver Geometry

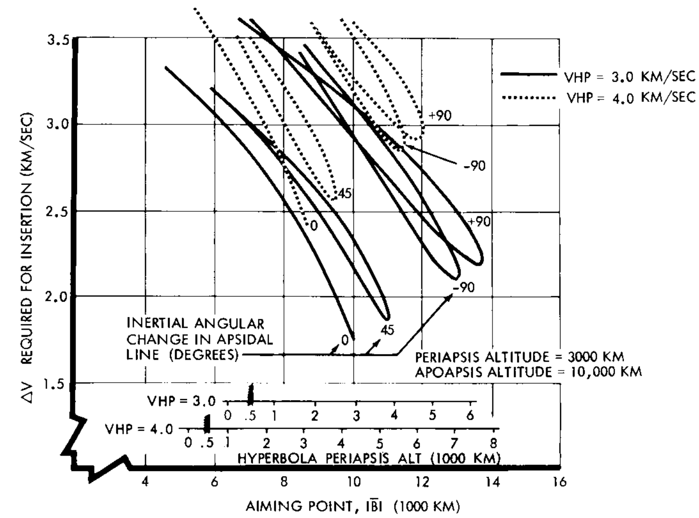


Figure 3.1-17: Insertion Velocity Requirements

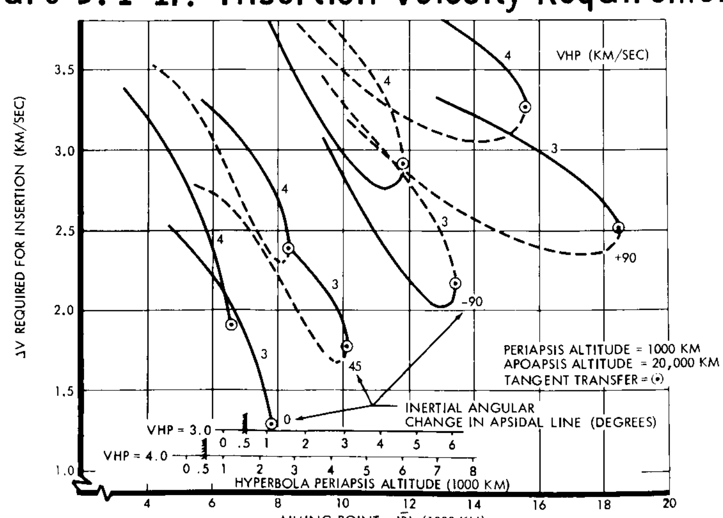


Figure 3.1-18: Insertion Velocity Requirements

the orbit, such as periapsis altitude and period, and those items that determine the orientation of the satellite orbit (longitude of ascending node, argument of periapsis, and inclination to the equator). The size and shape of the orbit may be selected preliminarily from the insertion engine performance, orbit lifetime, and the mission objectives. The orientation of the orbit involves the transit trajectory as well as interference with lines of sight to Canopus and Sun, communication to Earth, illumination of photographic regions, and ground coverage of the orbit.

Periapsis Altitude--Orbit lifetime often dictates the minimum periapsis altitude. Figure 3.1-19 shows the minimum size orbits that will maintain a 50-year orbit lifetime. This curve has been generated for the realistic atmosphere presented in Reference 1. The heavy solid curve for an M/C_DA of 0.27 is typical of the Boeing preferred design spacecraft. Periapsis altitudes as low as 300 km apparently can be used with the higher orbit periods, but solar gravity and orbit attainment inaccuracies will raise the number. Dispersions in the B-vector expected at encounter will require biasing the aiming point. This will result in a higher periapsis altitude, perhaps as high as 1000 km, to meet contamination constraints. Although the lowest possible orbit is usually preferred for observation of the planet surface, the Boeing Task A study showed that typical cameras without image-motion compensation would have the best resolution for orbits with periapsis altitudes from 1000 to 2000 km. Orbits with minimum periapsis altitudes of 1000 km have been chosen in the following representative examples. Higher periapsis altitudes may be obtained in most cases by a slight increase in ΔV requirements.

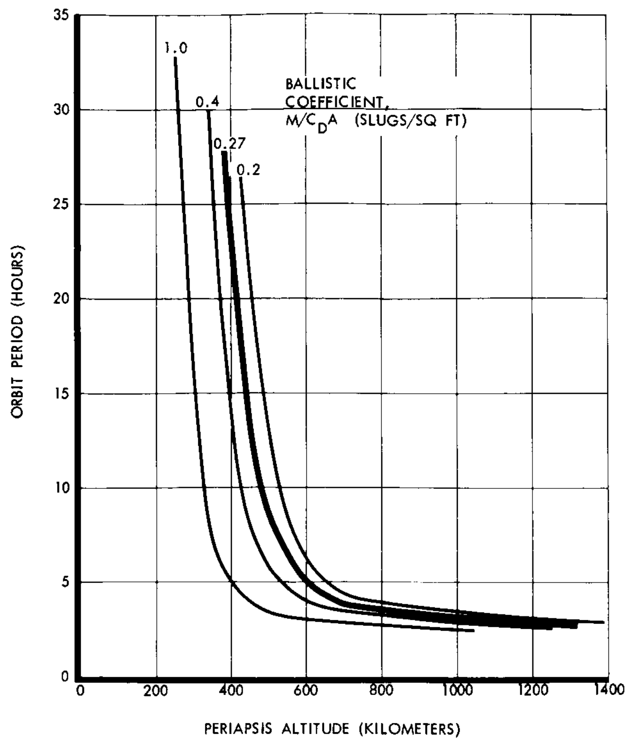


Figure 3.1-19: Minimum Orbit Size For A 50-Year Lifetime

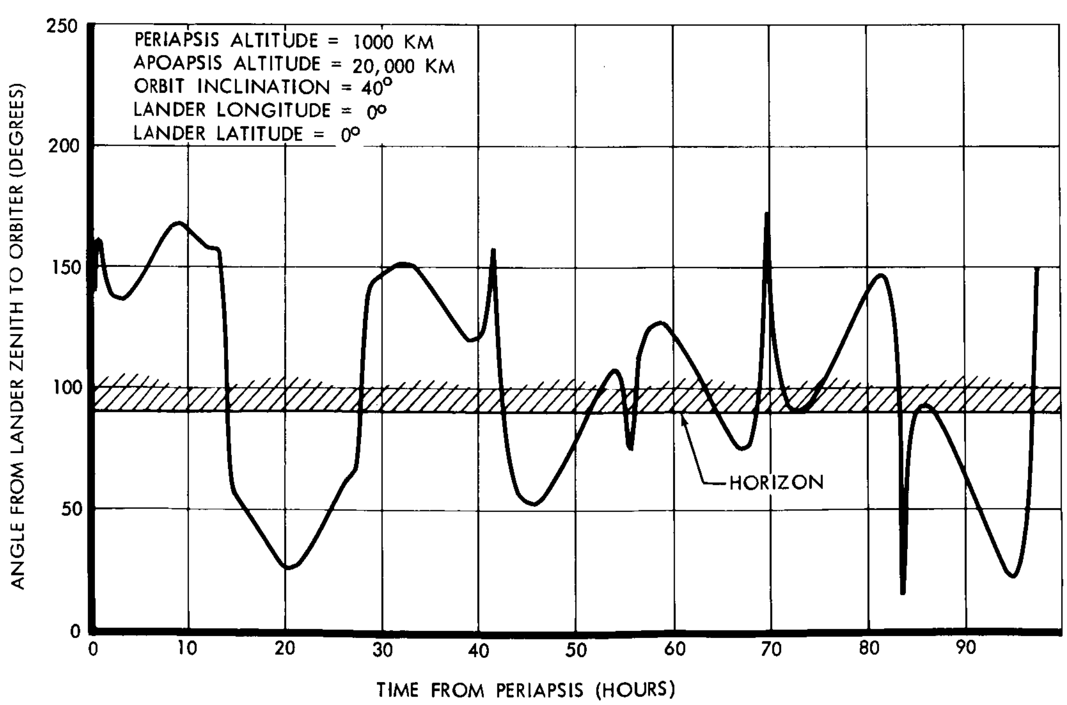


Figure 3.1-20: Orbiter/Lander Angular Relationship

Orbit Period--Propulsion requirements and orbit lifetime, as discussed in previous paragraphs, determine the minimum allowable orbital period. The maximum allowable orbital period is determined from the sensitivity of orbit period to the errors expected in the insertion maneuver and from the third-body effects caused by the Sun. Considering the above effects, the allowable maximum for a nominal orbital period is over 50 hours. This period is longer than required for science objectives, and a more reasonable upper value of 24 hours can be set. As the orbit period is decreased, the ΔV required of the insertion engine increases. Orbits with periods between 4 and 20 hours appear to meet most of the mission objectives, and these can be adjusted to give repetitive coverage of preferred target areas. For future Voyager missions, when communication between the landed capsule and the spacecraft becomes important, Boeing's preliminary studies show the strong effect of orbit period on the communication between the capsule and the spacecraft. Figure 3.1-20 shows the capsule-to-spacecraft angular relationship for a particular landing-site location of 0-degree aerographic longitude and 0-degree latitude. Example data for the first 5 days in orbit are shown. The angle shown on the vertical scale is measured from the capsule-local-vertical to the spacecraft. Angles greater than 90 degrees indicate that the spacecraft is beyond the horizon. A statistical approach similar to the method presented in Reference 3 can be applied to short-period orbits. For the longer orbit periods, especially those that approach synchronization with a specific surface feature, a different analysis is necessary to evaluate the available

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Reference 3: JPL Space Program Summary, No. 37-24, Vol. IV, June 1 to July 31, 1965.

communication time. Since maximum communication time is necessary during the early phase of the capsule mission, the effect of capsule-spacecraft communication on the selection of an orbit requires a detailed study.

Orbit Inclination--The orbit inclination is determined considering the desired ground coverage, illumination of specific surface features, interference with the Sun and Canopus sensors, and communication to the Earth. It is possible to obtain almost any orbit inclination. The only restriction is that the orbit inclination must be greater than the declination of the incoming-velocity asymptote, LVI (see Figure 3.1-7). Polar orbits are possible if desired, and periapsis can be adjusted to a variety of latitudes in either hemisphere. It should be noted that some of these orbits have a short interference time with the Canopus tracker. Although these polar orbits allow total planet mapping, some of the latitudes covered are not of special interest, the orbit may have its lowest altitude at far north latitudes, and the altitudes at the interesting latitudes may be too high. Orbits with inclinations near 45 degrees appear particularly attractive since solar interference is avoided for the early part of the mission and good Earth communication can be maintained. As the orbit inclination is increased, the length of time increases during which there is Canopus interference. As a result, a 35- to 50-degree inclination orbit provides a good compromise while still satisfying desired ground coverage.

Other Orbit Parameters--Two remaining orbit parameters depend in part upon the orientation of the approach asymptote. These last two orbit parameters are the longitude of the ascending node and the argument of periapsis. These parameters influence placement of the orbit plane and placement of the line of apsides. They affect mission parameters through influence upon the illumination angles, latitudes of flight, photographic resolution at various latitudes or longitudes, and ease of orbit determination.

Orbit Perturbations--The effect of the atmosphere on orbit lifetime was discussed in conjunction with Figure 3.1-19. The analysis used to obtain this figure was based on a nonrotating atmosphere with Mars as the only perturbing body. The effects of the Sun's gravitational attraction, solar radiation pressure, Mars oblateness, and a rotating atmosphere have been considered in a separate study. Two dynamic atmosphere models have been considered. One model assumes the atmosphere to be rotating at a constant angular velocity. The other model assumes a slipping effect wherein the atmosphere at the surface of Mars rotates with the surface and, at a distance from the planet, has zero inertial velocity. The three models below show a wide variation in the lifetime prediction for the same orbit.

Atmosphere Model	Predicted Lifetime (Years)
Stationary Atmosphere	49
Constant Angular Velocity	65
Slipping	55

The increase in lifetime for the constant-angular-velocity model is due to the atmosphere "pushing" the spacecraft at each apoapsis passage. Currently, the slipping atmosphere model is considered the most realistic. Solar gravity and radiation pressure do not cause the lifetime to decrease by 2 years for orbit periods up to 20 hours. Additional work could be done in this area to improve the dynamic atmosphere model. However, the results of a nonrotating atmosphere can be used as a proven conservative estimate for easterly-moving satellites.

For inclinations below 63.4 degrees, the oblateness of Mars will cause the longitude of the ascending node to regress and the argument of periapsis to progress. This movement of the orbit can be used to increase ground coverage or reduce interference with Sun, Earth, or Canopus. Figure 3.1-21 shows the angular rates expected for orbits with a periapsis altitude of 1000 km and periods ranging from 4 to 20 hours. The angular rotation rates are relatively high for short-period orbits with inclinations near 40 degrees. Note that a variation in $\dot{\omega}$ results in a corresponding change in both longitude and latitude, while $\dot{\Omega}$ affects only the longitude of a given suborbital point. The rates will usually be less for higher periapsis altitudes.

The long-term effects of the oblateness of Mars can be seen in Figure 3.1-22. The horizontal scale represents the argument of the spacecraft periapsis (angle measured from the ascending node to the spacecraft periapsis at any time), and the vertical scale represents the latitude. The solid lines shown trace the path of the spacecraft periapsis for three different inclinations. The dashed curves represent the traces of the two orbit points that have an altitude of 2000 km on a 1000-km periapsis altitude by 20,000-km apoapsis altitude orbit with a 40-degree inclination. This 2:1 change in altitude represents about a 2:1 change in the camera resolution. Allowed enough time, the spacecraft would eventually

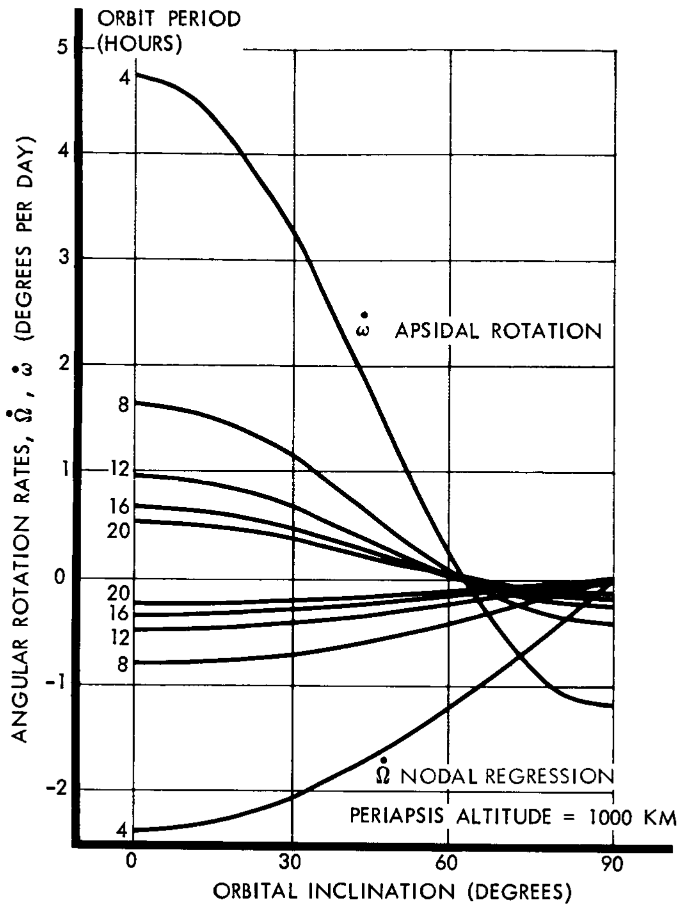


Figure 3.1-21: Angular Rotation Rates

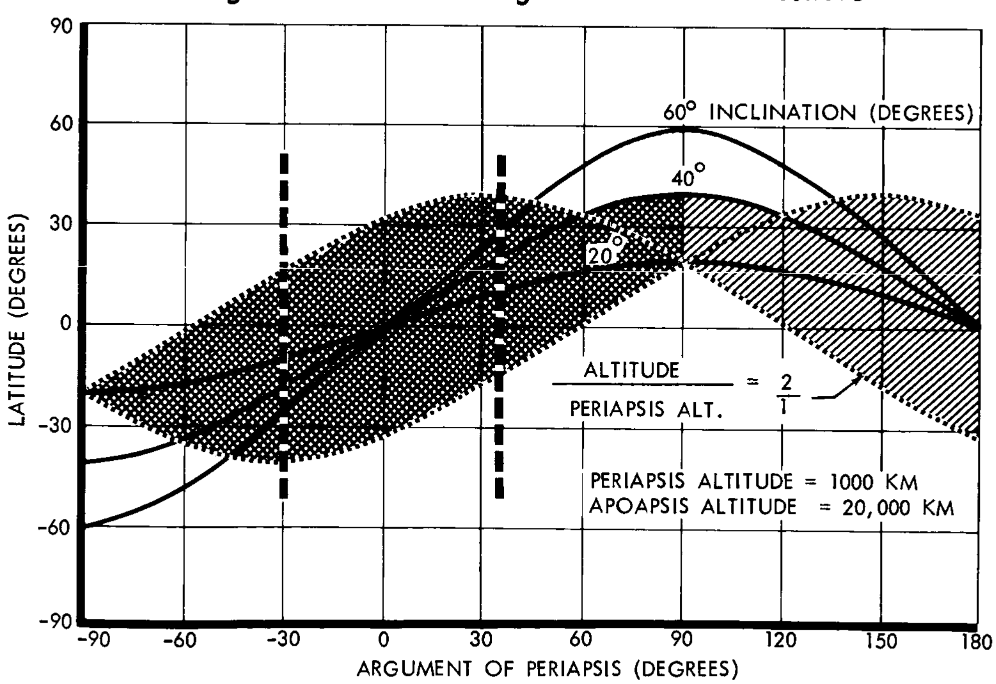


Figure 3.1-22: Long-Term Perturbation Effects

cover an entire cycle on this graph. The lightly shaded region represents the range of latitudes that would be covered during this period of time. A typical initial periapsis location is indicated by the lefthand heavy dashed vertical line. Over a 180-day period, the periapsis will progress to the righthand heavy dashed vertical line. The total region that would be covered within a 2:1 altitude range is indicated by the heavily shaded region. Higher inclinations of this orbit size will give more latitude coverage with narrower longitude bands.

The rotations can also be used to adjust the illumination angle as the apparent terminator moves. This can be accomplished by taking the vector sum of $\dot{\omega}$ and $\dot{\Omega}$ such that the Mars orbit net precession rate sums up to approximately 0.52-degree per day, the mean rotational rate of Mars around the Sun. If desired strongly enough to compromise other features, solar interference can be avoided during the entire mission with proper adjustment in the precession rate.

Characteristics of Typical Satellite Orbits--Figure 3.1-23 shows the latitude versus illumination-angle history for the first orbit at three different orbit inclinations. An S-vector corresponding to a typical launch and arrival date has been assumed for this curve. Different S-vectors will produce different curves, and this is shown only as a typical example. Insertion takes place at periapsis. The orbit size affects only the altitudes of the specific points that comprise the curve. The heavy band indicates the region that has a 2:1 change in the spacecraft altitude as referenced to periapsis. The blocked-off region indicates a

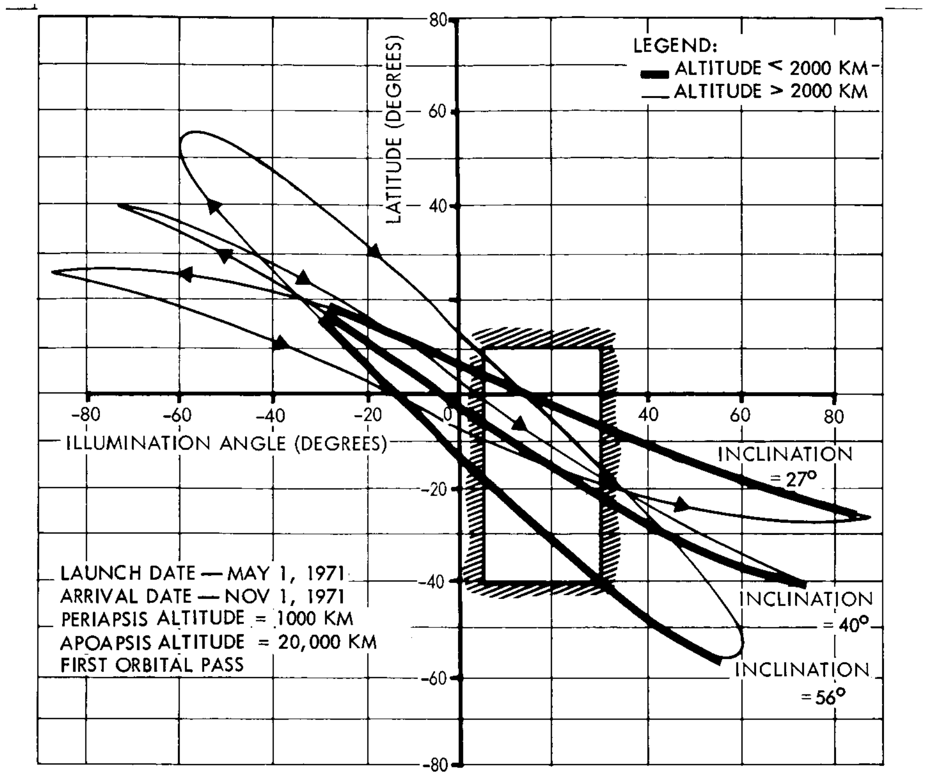


Figure 3.1-23: Latitude-Illumination Angle Variation

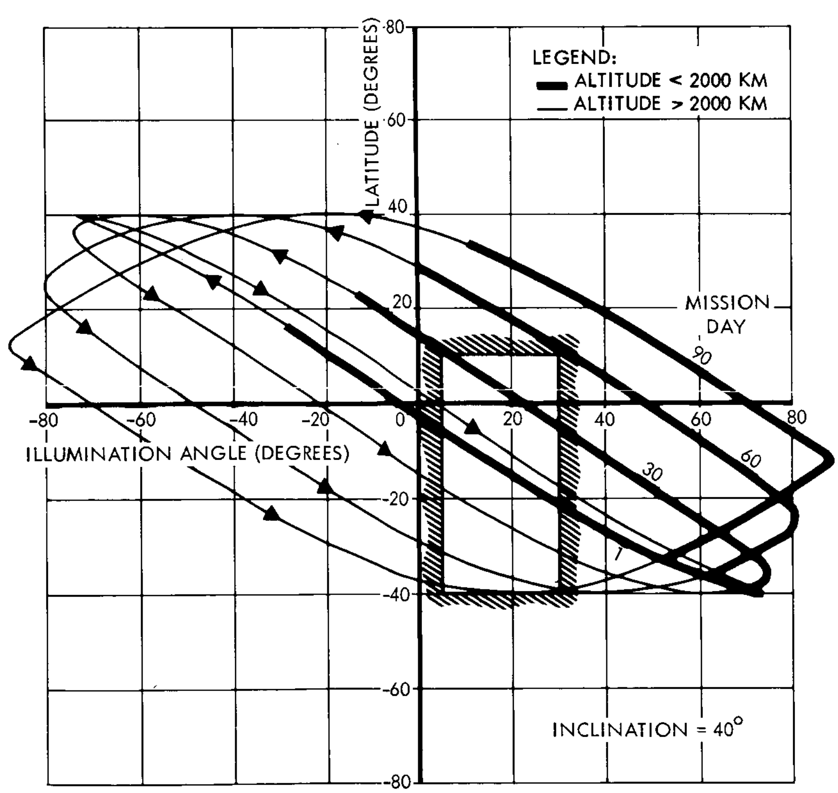


Figure 3.1-24: Latitude-Illumination Angle History

band of latitudes and illumination angles that appears desirable. The range of latitudes considered important is from +10 degrees to -40 degrees, and the range of illumination angles is from 5 to 30 degrees, measured from the terminator. All the orbits shown have good illumination angles and latitude coverage. Other curves would show that inclinations between 15 and 80 degrees appear to be satisfactory over the 2:1 range of altitudes. Adjustments of the line of apsides at insertion can improve the latitude and illumination angle history for many of the orbits that would otherwise be undesirable. This figure is for the first orbit about Mars. The same orbits at other days have different illumination angles and latitudes due to the motion of the Sun and the effects of oblateness of Mars. Figure 3.1-24 shows the effects of time upon the orbit. The 1st, 30th, 60th, and 90th days in orbit are indicated for a 40-degree inclination orbit. The example shown has excellent illumination angles and latitude coverage for the first 60 days of the mission and good coverage thereafter.

All of the orbits shown have some degree of Mars interference with the Canopus tracker. Interference with the Canopus tracker is assumed to occur when Mars appears in an area defined by a cone angle of 90 $\pm$60 degrees and a clock angle of $\pm$35 degrees. Those orbits which have Canopus interference for greater than 4 hours have been discarded in this preliminary analysis. Orbits that give interference with Sun, Earth, or Canopus prior to the orbit-insertion maneuver have also been discarded. It is believed that the time just prior to insertion is critical, and any loss of a reference body or communication will seriously affect the probability of attaining a satisfactory orbit about Mars. Most orbits with inclinations between 30 and 115 degrees meet these two requirements.

Figure 3.1-25 shows the time that spacecraft spends in the solar umbra. Orbits with inclinations greater than 45 degrees appear to be promising since they are completely free from occultation during the first 60 days of the mission. However, a large percentage of orbits with inclinations greater than 45 degrees have Canopus interference for more than 4 hours, and, as a result, are unacceptable. The Voyager Flight Spacecraft can safely be inserted into a wide range of orbits with periods as low as 2.8 hours and corresponding periapsis altitudes as low as 1000 km. Figure 3.1-26 shows the maximum occultation times that can be encountered. Occultation is assumed to occur when the satellite orbit intersects a cylinder projected from Mars. The cylinder has its central axis along the negative projection of a vector leading to the viewed object. When the spacecraft is in this cylindrical region, total occultation is assumed to occur. This very closely represents the umbra for the case of solar occultation. The maximum possible occultation will occur when the apoapsis of the satellite orbit is on the central axis of the cylinder. The dashed curves on Figure 3.1-26 show the maximum occultation time per orbit possible for the satellite orbits under consideration. The solid curves show the maximum occulted period as a percentage of the orbit period. The limit shown at the bottom represents the minimum orbit size that is available to provide a 50-year lifetime with an M/C_DA of 0.27 slug/ft^2. Figure 3.1-27 shows a cone-clock angle plot for Mars and the two limbs during the first orbit. Only those areas that are not shaded are acceptable for the sensor vision. Interference with the Canopus sensor occurs for only 2.5 hours in this 40-degree orbit because interference occurs near periapsis. Interference with communication to the Earth is

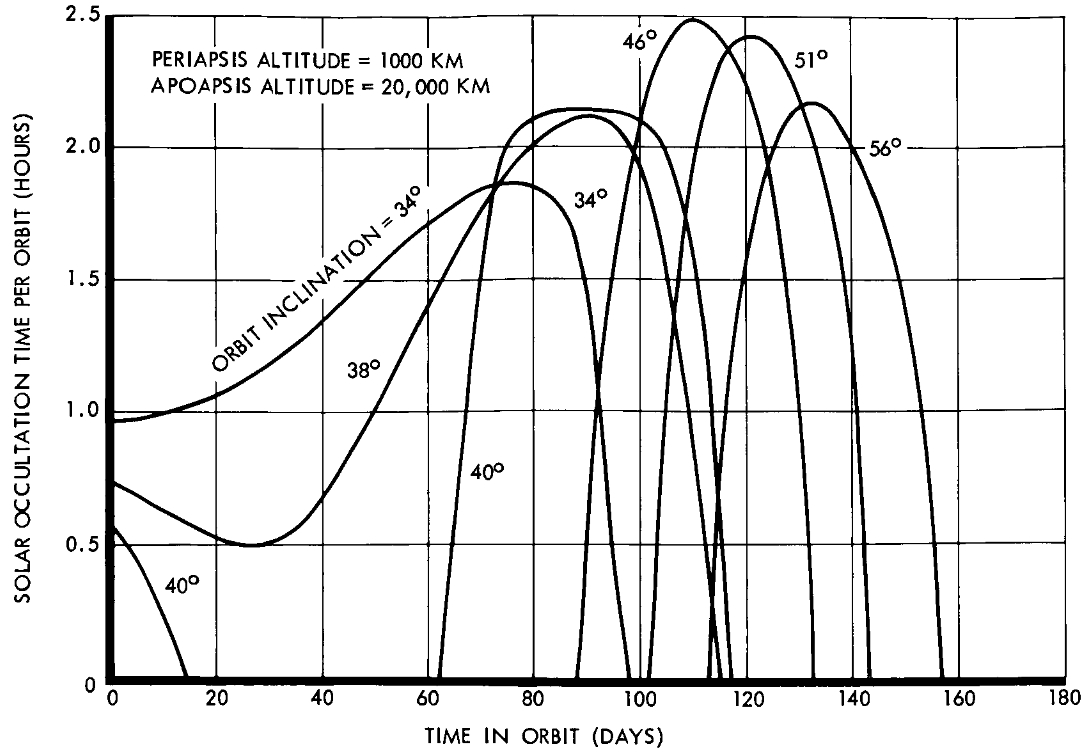


Figure 3.1-25: Solar Occultation in Orbit

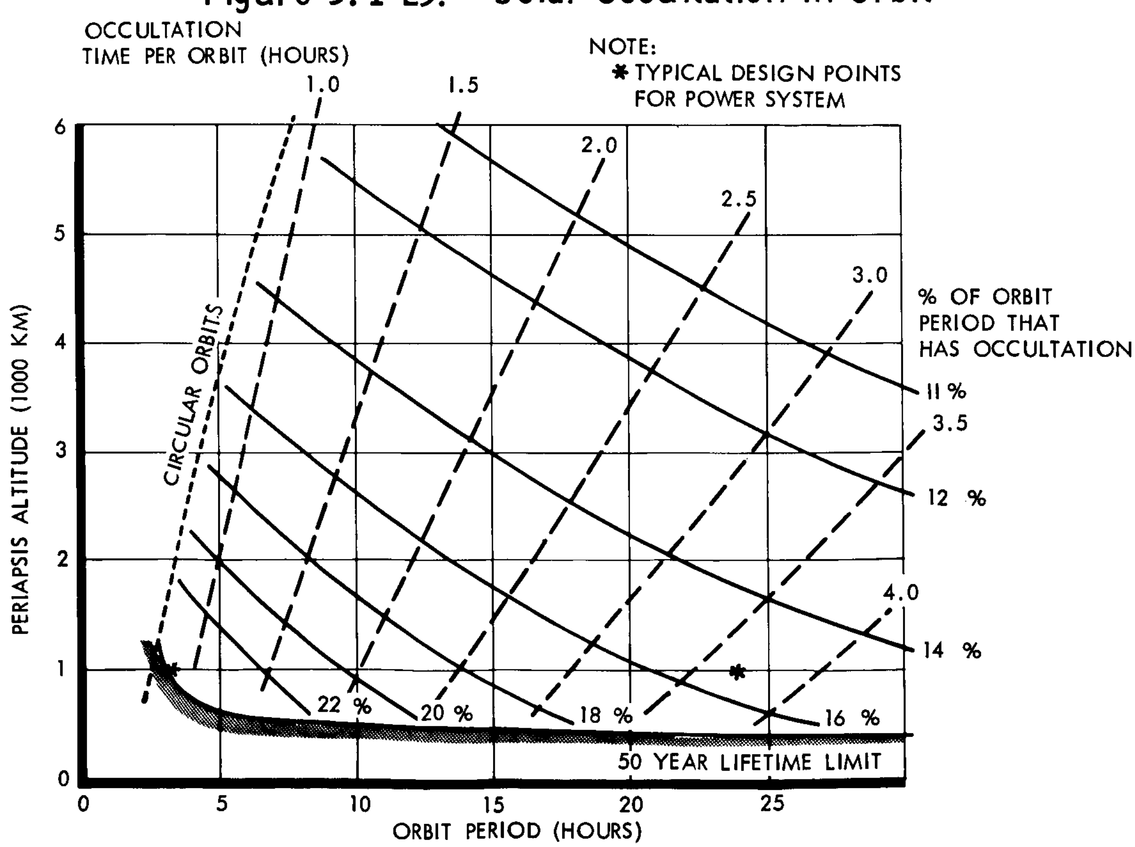


Figure 3.1-26: Maximum Possible Occultation in Orbit

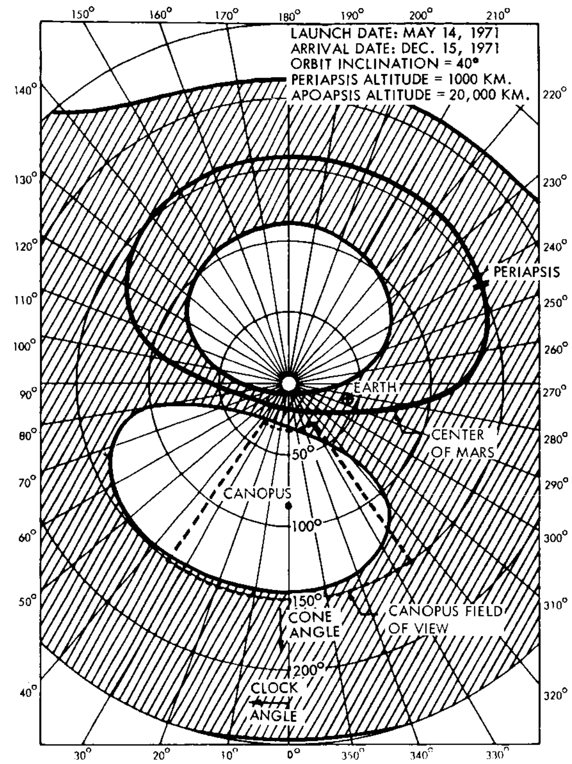


Figure 3.1-27: Cone and Clock Angles in Orbit

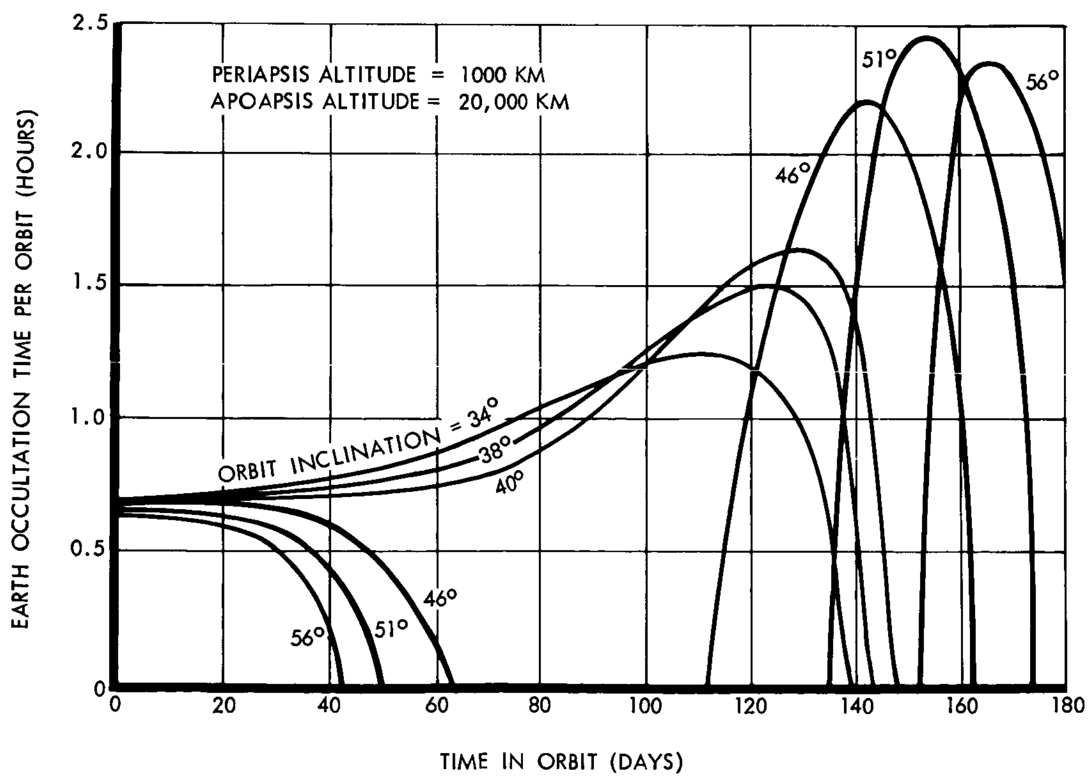


Figure 3.1-28: Earth Occultation in Orbit

shown by Figure 3.1-28. All of the orbits shown will have some loss of communication during the mission, but the percentage is small. Note that small changes in the inclination do not affect communications during the critical first 20 days of the mission.

Competing Objects--The two moons, Phobos and Deimos, in orbit around Mars provide a potential hazard to the spacecraft. In addition, any collision that might take place in a 50-year period of time could result in an unsterilized particle impacting the planet. The problem of impacting Phobos and Deimos during the first 50 years in orbit has been investigated for a variety of orbits. The results are presented below.

PHOBOS AND DEIMOS 50-YEAR
IMPACT PROBABILITIES

Periapsis Radius (km)	Apoapsis Radius (km)	Orbit Inclination	Phobos-Impact Probability	Deimos-Impact Probability
4400	23505	40°	5.7×10^{-6}	3.8×10^{-6}
4400	9360	40°	1.5×10^{-4}	0
9340	9360	40°	1.6×10^{-5}	0
4400	23505	0°	8.3×10^{-6}	4.3×10^{-6}
4400	9360	0°	1.3×10^{-1}	0
9340	9360	0°	0.999	0

Orbits with high inclinations have a relatively low probability of impacting Phobos and Deimos. For low inclinations, the probability of impact increases considerably as the orbit period approaches the period of either of the moons.

3.1.4 Flight Capsule Entry Trajectory

Deflection Geometry--The basic geometry involved in the deflection to entry maneuvers is shown in Figure 3.1-29. At a point along the satellite orbit, the Flight Capsule and Spacecraft are separated. The impulse applied at the separation maneuver will be sufficiently large to achieve a 1000-meter separation distance in 20 minutes. At this time, the deorbit motor will be ignited to impart a ΔV and establish the orbital descent trajectory. The magnitude of this ΔV is dependent upon where in the orbit the maneuver is performed, the entry angle, and the direction in which the deorbit motor is fired. The location of the deflection maneuver is measured by its true anomaly (angle from periapsis to deflection point measured in the direction of motion). The firing angle is referenced to the entry angle of attack (angle between velocity vector at entry and thrust axis).

ΔV Requirements for Deorbit--Figure 3.1-30 shows the ΔV requirements to deflect the capsule on an entry trajectory as a function of the deflection location (true anomaly). The capsule skip-out limit for a ballistic coefficient, M/C_DA, of ∞ is shown as well as a maximum allowable entry angle of 20 degrees. The angle-of-attack limit of 60 degrees and the ΔV limit of 550 meters per second narrow the range of allowable descent trajectories. There is an additional constraint that may be used to narrow the range of available capsule descent trajectories. This constraint places the landing site location 15 to 30 degrees above the terminator. Since desirable satellite orbits have their periapsis at about the same location relative to the terminator, the capsule descent trajectory and satellite orbit can be fixed in inertial space.

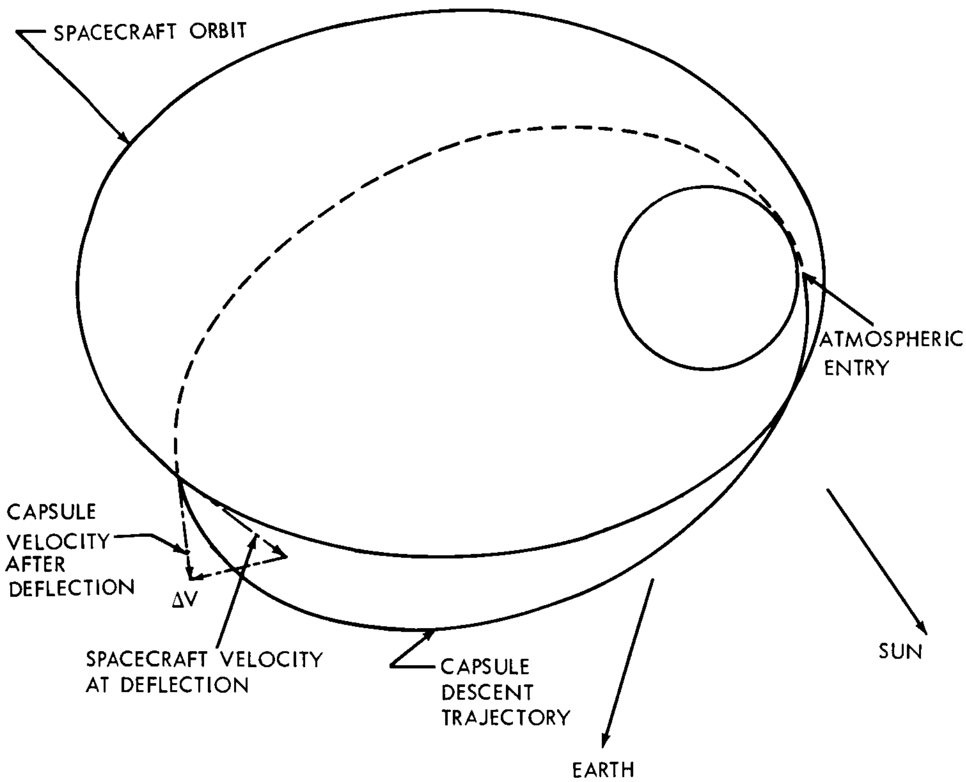


Figure 3.1-29: Deflection Geometry

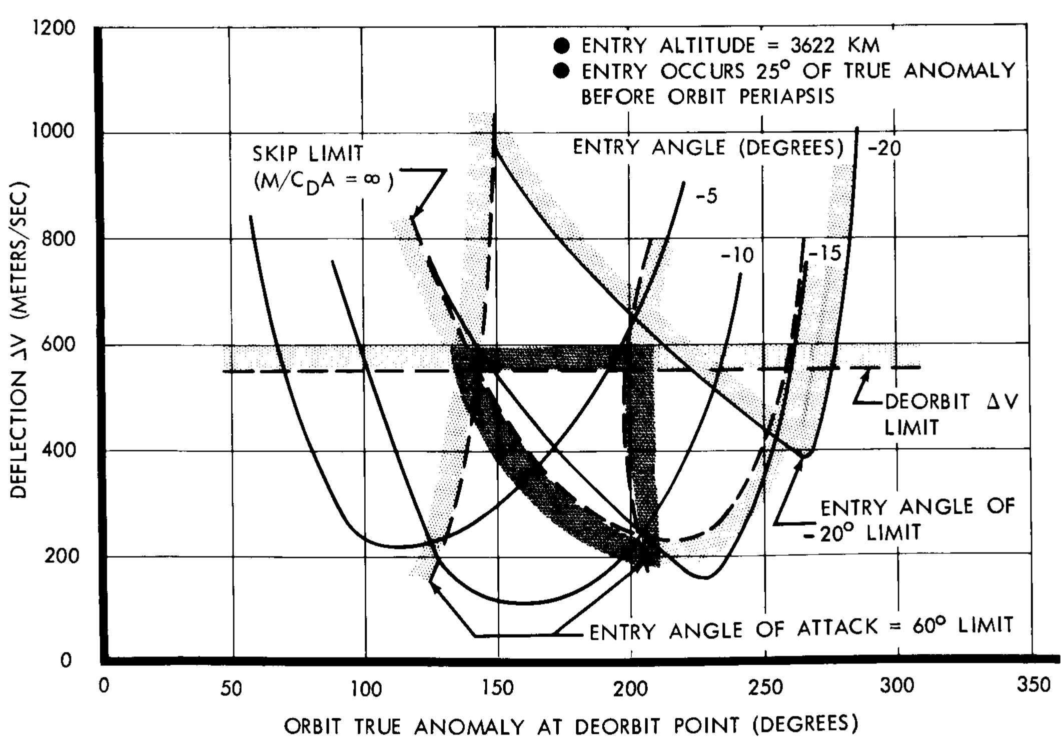


Figure 3.1-30: Capsule Trajectory Constraints

Communication Considerations--It is desirable to maintain communication between the spacecraft and the Earth during the entire descent phase. Communication can be maintained by selection of deflection-true anomalies outside the range from 28 to 102 degrees. An additional desire to be in view of Goldstone during the entry phase affects the timing of the entire mission. This timing may be controlled by adjustments in the orbital period, time of separation, and arrival of the Flight Spacecraft.

3.2 ORBIT/ENTRY DETERMINATION UNCERTAINTIES

3.2.1 Initial Mars Orbit Determination

Estimated orbit determination errors versus number of periapsis passages are summarized below for a 13.8-hour orbit inclined at 40 degrees to the Mars equator with argument of periapsis at 10 degrees from the ascending node. Studied data combinations were doppler only, doppler supplemented with range marks once per hour, and doppler supplemented with range or with measurements of the direction of Mars local vertical. The 1-sigma data errors assumed were 0.001 meter/second in range rate derived from doppler, 0.005 km in range, and 0.2 degree in each of two angles fixing the direction of local vertical. The dominant factor in early orbit determination is the validity of the model to which the data are fitted. Given a good model, the dominant error is in the orientation of the orbit plane about the line of sight from Earth. Using doppler data only, the position at periapsis is determined to within 4 km after the first periapsis passage and 0.4 km after the second periapsis passage. Accuracies are not improved significantly with the addition of range or angle data; however, the convergence rate is approximately the same with range data only as with doppler data only. Therefore, the range data may be useful if ranging is established before high-accuracy doppler tracking. Results are summarized in the following table.

	Doppler Only		Doppler Plus Range		Doppler Plus Angle	
Number of Periapsis Passages	1	2	1	2	1	2
Oblateness Constant (Percent)	0.1	0.01	0.1	0.01	0.1	0.01
Mars-Earth Relative Velocity (Meters per Second)	0.001	0.0001	0.001	0.0001	0.001	0.0001
Ascending Node in Plane of Sky (Degree)	0.06	0.01	0.05	0.01	0.02	0.01
Position at Periapsis						
Vertical (km)	1.0	0.1	0.8	0.1	1.0	0.1
Crossplane (km)	2.5	0.2	2.2	0.15	1.0	0.2
Downrange (km)	2.0	0.2	2.0	0.2	2.0	0.2

3.2.2 Flight Capsule Entry and Impact Conditions

The Flight Capsule orbit parameter uncertainties at atmospheric entry and impact were estimated using combinations of range rate measured from the orbiter, local vertical measured on the capsule, and altitude measured on the capsule. Observation 1-sigma errors of 0.1 meter per second, 0.5 degree, and 0.1 km, respectively, were assumed. Using range-rate data only, the 1-sigma uncertainty in the prediction of entry velocity is about 0.2 meter per second; adding altitude measurements, the uncertainties are reduced to about 0.05 meter per second. Atmospheric parameters can be estimated from the difference between predicted and measured velocity in the line of sight when the effect of atmosphere on the measured component is 1 meter per second or greater. The crossplane impact uncertainty is 4 km. This is not reduced significantly by angle measurements on the capsule with 1-sigma errors of 0.25 degree.

3.3 VOYAGER FLIGHT SPACECRAFT COMPONENTS DESIGN PARAMETERS

A summary of weights and reliability assessments for major items of the Planetary Vehicle is presented as Table 3.3-1.

Table 3.3-2 presents component design parameters for the preferred Spacecraft Bus, Propulsion Subsystem, Planetary Vehicle Adapter, and Science Subsystem. The table has been simplified in keeping with the level of detail appropriate to Task B. More complete data will be prepared and tabulated during Phase IB and Phase II.

Table 3.3-1: SUMMARY OF PLANETARY VEHICLE CHARACTERISTICS

SYSTEM ELEMENT	WEIGHT		RELIABILITY ALLOC.	RELIABILITY ASSESS.
SPACECRAFT BUS		2,100		
Power	541.6		0.980	0.9932
Guidance & Control	266.4		0.994	0.9945
Data Storage	107.7		0.940	0.9490
Telemetry	57.3		0.990	0.9978
Radio	162.5		0.970	0.9820
Command	31.4		0.970	0.9847
Computing & Sequencing	45.0		0.970	0.9860
Structural & Mechanical	408.5		0.998	0.9984
Pyrotechnic	34.0		0.999	0.9999
Temperature Control	80.1		0.996*	0.9990*
Cabling	184.0		0.999	0.9960
Contingency (weight)	181.5		---	---
PROPULSION SUBSYSTEM		15,000	0.995	0.9960
Midcourse & Orbit Trim Propulsion	3,962.0			
Orbit Insertion Propulsion	10,400.0			
Cabling & Power Conditioning	25.0			
Structures & Mechanical	337.0			
Temperature Control	118.0			
Contingency (weight)	158.0			
SCIENCE SUBSYSTEM		400	0.800	0.8001
FLIGHT SPACECRAFT TOTAL		(17,500)	0.621	0.7202
CAPSULE SYSTEM		2,000	---	---
Capsule & Sterilization Canister	1,968.0			
Radio Link & Data Storage	32.0			
PLANETARY VEHICLE TOTAL		(19,500)	---	---
PLANETARY VEHICLE ADAPTER		850	0.999	0.9990
PLANETARY VEHICLE & ADAPTER TOTAL		(20,350)	---	---

* Includes Propulsion Temperature Control Reliability

TABLE 3.3-2: COMPONENT DESIGN PARAMETERS

ITEMS * EQUIPMENT TOTALS MODIFIED BY DUTY CYCLE	QTY PER S/C	CURRENT WT. EST. ITEM	CURRENT WT. EST. TOTAL (SUBTOTAL)	ESTIMATED HEAT DISSIPATION WATTS*	ALLOCATED POWER WATTS	RELIABILITY ASSESSMENT
SPACECRAFT BUS			2100		205	.9045
POWER SUBSYSTEM			((541.6))			.9932
Solar Panel - Fixed	(8)	(18.8)	(150.2)			
Substrate Assy.	1	13.6				
Support Frame and Struts	1 set	3.65				
Fittings and Attachments	1 set	.22				
Thermal Coating	A/R	1.34				
Solar Panel - Deployable	(4)	(30.5)	(121.9)			
Substrate Assy.	1	18.1				
Support Frame and Struts	1 set	3.1				
Thermal Coating & Dielectric	A/R	2.1				
Deployment Mechanism	1	1.75				
Deploy Latches	1 pr	2.7				
Stowage Latches	1	2.75				
Battery, Regulator & Inverter Assy (Bay 8)	(1)		(80.4)			
Battery	1	49.5		58		
Battery Charger	1	4.6		8		
DC Regulator	1	6.7		26		
Fail Sense	1	2.0		1		
2400 cps Inverter	1	5.1		26		
Chassis, Radiator & Hardware	1	11.0				
Wiring and Connectors	A/R	1.5				
Battery, Regulator & Inverter Assy (Bay 7)	1	80.4	(80.4)			
Same as Assy. 8 above						
Battery, Regulator & Inverter Assy (Bay 5)	1	80.4	(80.4)			
Same as Assy. 8 above						
Power Control Assy (3/4 of Bay 6)	(1)		(28.3)			
Power Switching and Logic	1	6.3		56		
Booster Converter & Share Sensor	1	2.1		1		
400 ∿ Inverter	2	4.2		2		
Power Control Sub Assy	1	3.1		0		
Precision Oscillator Sub Assy	1	4.6		4		
Chassis, Radiator, & Hardware	1	6.8				
Wiring & Connectors	A/R	1.2				
RADIO SUBSYSTEM			((162.5))		172.7	.9820
Radio Electronics Assy (Bay 3)	(1)		(54.2)			
Power Amplifier Subassembly						
Power Amplifier & Power Converter	2	20		100		
Band Reject Filter	2	1.6				
Circulator 4 Port	2	4.8		.12		
Circulator 3 Port	1	0.8		.03		
Hybrid	1	1.				
Relay Receiver Subassembly						
Receiver (GFE)	2	Ref. (9.2)		10		
Power Converter (GFE)	2	Ref. (5.0)		3		
Chassis, Radiator & Hardware	1	23				
Wiring & Connectors	A/R	3				
Radio Electronics Assy (Bay 2)	(1)		(81.3)			
S-Band Radio Subassembly						
Receiver	2	11		5.6		
Power Converter	2	5		2.1		
Ranging Unit	2	9		5.0		
Preamplifier	2	3		0.3		
Diplexer	2	3				
Preselector	2	1.6				
Exciter Subassembly						
Exciter	2	4		5.4		
Power Converter	2	5		1.8		
Isolator	2	2.4				
Band Pass Filter	2	2				
Launch Transmitter Subassembly						
Launch Transmitter	1	2.5		28		
Power Converter	1	4		6		
Isolator	1	0.8				
Band Pass Filter	1	1				
Redundancy Control Subassembly						
Redundancy Control S/A	1	8		2		
Power Converter	1	3.0		1.1		
Chassis, Radiator & Hardware	1	11				
Wiring & Connectors	1	5.0				
Antennas	(1)		(27.0)			
High Gain Antenna	1	12.5				
Med. Gain Antenna	1	5.0				
Low Gain Antenna and Boom	2	8.0				
VHF Relay Antenna	1	1.5				

TABLE 3.3-2: COMPONENT DESIGN PARAMETERS (CONTINUED)

ITEMS * EQUIPMENT TOTALS MODIFIED BY DUTY CYCLE	QTY PER S/C	CURRENT WT. EST. ITEM	CURRENT WT. EST. TOTAL (SUBTOTAL)	ESTIMATED HEAT DISSIPATION WATTS*	ALLOCATED POWER WATTS	RELI-ABILITY ASSESS-MENT
SPACECRAFT BUS (continued)						
COMMAND SUBSYSTEM			((31.4))		20	.9847
Electronic Assembly (½ of Bay 16)	1		(31.4)			
Command Detector Subassembly	1	10.0		3.0		
Command Decoder Subassembly	1	7.6		11.2		
Command Output Combiner	1	1.0		0		
Power Conditioner Subassembly	1	4.4		1.6		
Chassis, Radiator & Hardware	1	5.5				
Wiring and Connectors	A/R	2.9				
DATA STORAGE SUBSYSTEM			((107.7))		50	.9490
Data Storage Assy (Bay 14)	(1)		(48.4)			
TV Recorder #1	1	18		12		
TV Recorder #2	1	18		12		
Capsule Recorder	1	Ref (18)		12		
Chassis, Radiator, & Hardware	1	11.0				
Wiring & Connectors	A/R	1.4				
Data Storage Assy (½ of Bay 16)	(1)		(59.3)			
IR Scanner Recorder	1	12		6		
IR - UV Spectrometer Recorder	1	12		6		
Field & Particles Recorder	1	12		6		
Maneuver Recorder	1	12		6		
Interface Unit (Programmer)	1	3				
Chassis, Radiator & Hardware	1	5.5				
Wiring & Connectors	A/R	2.8				
TELEMETRY SUBSYSTEM (Bay 4)			((57.3))		7.5	.9978
Electronic Assembly	(1)		(57.3)			
Power Converter Subassembly		2.4				
Master Digital Multiplexer Subassembly		2.0				
Engineering Multiplexer Subassembly		13.24				
Block Encoder Subassembly		0.2				
Upper Subcarrier Bi-Phase Modulator Subassembly		0.5		13.4		
Base Band Generator Subassembly		5.46				
Data Buffer Subassembly		2.5				
Signal Conditioning & Junction Box Subassembly	1	8		2.5		
Chassis, Radiator & Hardware	1	11				
Wiring & Connectors		12				
GUIDANCE AND CONTROL SUBSYSTEM			((266.4))		174	.9945
Attitude Reference & Autopilot Assy (Bay 1)	(1)		(74.6)			
Inertial Reference Unit	1	30		33		
Canopus Sensor	2	10		8		
Fine Sun Sensor	1	1		3		
Autopilot Electronics	1	9		61		
Attitude Ref. Electronics	1	11		56		
Chassis, Radiator & Hardware	1	11.5		0		
Wiring & Connectors	A/R	2.1				
Guidance Platform Assembly	(1)		(10.0)			
Planet Tracker	1	2.0		1		
Platform, Cover & Boom	1	3.0				
Motor Drive & Gimbal	1	5.0		16		
Coarse Sun Sensor Assembly	(4)	.5 ea.	(2.0)	.05		
Reaction Control Assembly	(2)		(165.0)			
Nitrogen Tank (incl. Temp. Trans)	4	96				
Thrusters	16	10		56		
Valves, Regulators & Press. Trans.	2 sets	4				
Tubing & Instl. Hardware	2 sets	11				
Reaction Control Gas (N_2)		44				
Antenna Pointing Control Assembly	1	14	(14)	16		
Limb/Terminator Sensor Assembly	2	.5	(0.5)	1		
Earth Sensor	1	0.3	(0.3)	1		

TABLE 3.3-2: COMPONENT DESIGN PARAMETERS (CONTINUED)

ITEMS * EQUIPMENT TOTALS MODIFIED BY DUTY CYCLE	QTY PER S/C	CURRENT WT. EST. ITEM	CURRENT WT. EST. TOTAL (SUBTOTAL)	ESTIMATED HEAT DISSIPATION WATTS*	ALLOCATED POWER WATTS	RELI-ABILITY ASSESS-MENT
SPACECRAFT BUS (continued)						
STRUCTURAL & MECHANICAL SUBSYSTEM			((408.5))			.9984
Primary Structure			(218)			
Skin & Doublers		87.2				
Stiffeners	48	37.8				
Longerons	8	16.0				
Rings	4	62.0				
Misc. Fittings & Fasteners	A/R	15.0				
Secondary Structure			(140.5)			
Meteoroid Shielding		52.7				
Solar Panel Supports	8	5.0				
Antenna Supports		19.5				
Science System Supports		35.0				
Guidance Platform Support		2.0				
Reaction Control Tank Supports	4	12.0				
Reaction Control Thruster Supports	4	2.0				
Misc. Brackets and Fasteners		12.3				
Capsule Emergency Separation Provisions			(25)			
Capslle Support Field Joint Ring		15.0				
V Block Band		6.0				
Emergency Separation Mechanism		4.0				
Planetary Vehicle Separation Provision						
Support Ring		Ref. (15)				
Block Band - Clamp		Ref. (10)				
Separation Mechanism		Ref. (25)				
Deployment Mechanisms			(25)			
High Gain Ant. Boom Latch & Deploy Mechanism	1 set	15				
Medium Gain Ant. Latch & Deploy Mechanism	1 set	3				
Low Gain Ant. Latch & Deploy Mechanism	2 sets	6				
Guidance Platform Latching Mechanism		1.0				
COMPUTING AND SEQUENCING SUBSYSTEM			((45.0))		70	.9860
C&S Electronic Assy (5/8 of Bay 15)			(45.0)	52		
Command Register Subassembly 8"	2	(1.10)				
Instruction Register Subassembly 8"	2	(1.10)				
Velocity Conversion Register Subassy.	1	.55				
Countdown Chain Subassembly 8"	1	.55				
Arithmetic & Memory Control Subassy.	1	.55				
Interrupt Register Subassembly	1	.55				
Memory Subassembly	2	(8.00)				
Transformer Subassembly	1	2.08				
Signal Conditioning Subassembly	3	(4.80)				
Output Matrix Decoder Subassembly	6	(3.30)				
Control Register & Timing Subassembly	2	(1.10)				
Power Conditioning Subassembly	2	(8.12)				
Chassis, Radiator & Hardware	1	6.90				
Wiring and Connectors		6.30				
PYROTECHNIC SUBSYSTEM			((34.0))		12	.9999
Pyrotechnic Power Switching Assy (3/8 of Bay 15)			(30.0)			
Capacitor Bank Subassembly	2	(4.24)		7.5		
Power Conditioning Subassembly	2	(5.56)				
Gates and Drivers Subassembly	8	(12.60)				
Chassis, Radiator & Hardware	1	4.10				
Wiring and Connectors	A/R	3.50				
Electro-Explosive Devices (EED's)	59		(3.0)			
Pyrotechnic Arming Switch (PAS)	1		(0.3)			
Separation - Initiated Timer (SIT)	1		(0.7)			
TEMPERATURE CONTROL SUBSYSTEM			((80.1))		40	.9990
Louver Assys	13	39.0				
Thermal Coating (Radiator Plates & Thermal Tie)		6.85				
Thermal Coating - (Electronic Assys)		3.0				
Heater and Switches	8/16	2.46				
Thermal Coatings - Ext. Mounted Equip.		2.0				
Insulation - Electronic Assemblies		9.4				
Electronic Assemblies - Thermal Ties	A/R	17.4				
CABLING			((184.0))			.9960
Cabling Ring Harness (Wire, Connectors, and Clamps)	1 set		(184.0)			
CONTINGENCY			((181.5))			

TABLE 3.3-2: COMPONENT DESIGN PARAMETERS (CONTINUED)

ITEMS * EQUIPMENT TOTALS MODIFIED BY DUTY CYCLE	QTY PER S/C	CURRENT WT. EST. ITEM	CURRENT WT. EST. TOTAL (SUBTOTAL)	ESTIMATED HEAT DISSIPATION WATTS*	ALLOCATED POWER WATTS	RELI-ABILITY ASSESS-MENT
PLANETARY VEHICLE ADAPTER						
STRUCTURAL & MECHANICAL SUBSYSTEM			((850))			.9990
Adapter Structure	1	730	(790)			
Cooling Air Duct	1	10				
Planetary Vehicle Separation Provision (Ref. Spacecraft Bus - Structural & Mechanical Subsystem)		50				
CABLING			(20)			
Adapter Cables, Conn., Hdwr. & EED's	1 set	20				
Inflight Destruct Provision		20	(20)			
Instrumentation Provision		20	(20)			
PROPULSION SUBSYSTEM			((15,000))		1 (contin.)	.9960
MIDCOURSE/ORBIT TRIM PROPULSION			(3962)		750 (15 ms)	
Monopropellant Engine (incl. jet vanes)	4	(87)				
Propellant Tank	4	(263)				
Propellant Feed		17				
Pressurant Tank		193				
Pressurant Feed		19				
Pressurant Gas (N_2) (incl. residual)		89				
Usable Propellant (Hydrazine)		3190				
Residual Propellant		104				
ORBIT INSERTION PROPULSION			(10,400)			
Rocket Motor Inerts		965				
Thrust Vector Controls		130				
Pressurant Feed		37				
Roll Control Jet & Control Valve Set		(5)				
Freon (liquid) (incl. residual)		203				
Pressurization Gas (N_2) (incl. residual)		15				
Usable Solid Propellant		9045				
CABLING & POWER CONDITIONING			(25.0)	1 (continuous)		
Converters & Switch Installation		11.4		750 (15 ms		
Cable Harness		13.6				
STRUCTURE			(337)			
Primary Support Frame		116				
Hydrazine Tank Support		12				
Nitrogen Tank Support		3				
Solid Motor Support		64				
Midcourse Engine Thrust Structure		27				
Meteoroid Shielding - Bus - Capsule		53				
Meteoroid Shielding - Solar Shield		51				
Miscellaneous Support Structure		11				
TEMPERATURE CONTROL			(118)			
Louver Assembly (installed on Bus)	5	(6.1)				
Radiator Plate (installed on Bus)	5	(3.9)				
Insulation (partly on Bus)	A/R	43.1				
Coatings (partly on Bus)	A/R	16.6				
Electrical Heaters & Switches	A/R	3.3				
Solar Heat Shields (set)	1	45				
CONTINGENCY			(158)			
SCIENCE SYSTEM (GFE)			((400))		160	.8001
Science Instruments (Platform Mounted)		100.0				
Science Instruments (Body & External Mounted)		140.0				
Data Automation Equipment		47.0				
Scan Platform No. I		53.5				
Platform No. II (UV Spectrometer)		14.0				
Booms and Antennas		28.0				
Power Switching Electronics		6.0				
Wire Harness and Miscellaneous		11.5				
Note: Science Instrument Remote Electronics and DAE are located in Bays 11 and 12. UV spectrometer is located in Bay 13.						

* Equipment totals modified by duty cycle

3.4 VOYAGER EQUIPMENT ELEMENTS AND DOCUMENTATION IDENTIFICATION

The method for identifying elements of Voyager Spacecraft System equipment, as described in the Task A formal report, is still fully applicable. The equipment-list drawing tree (Figure 3.4-1) has been revised to reflect the Task B preferred spacecraft configuration.

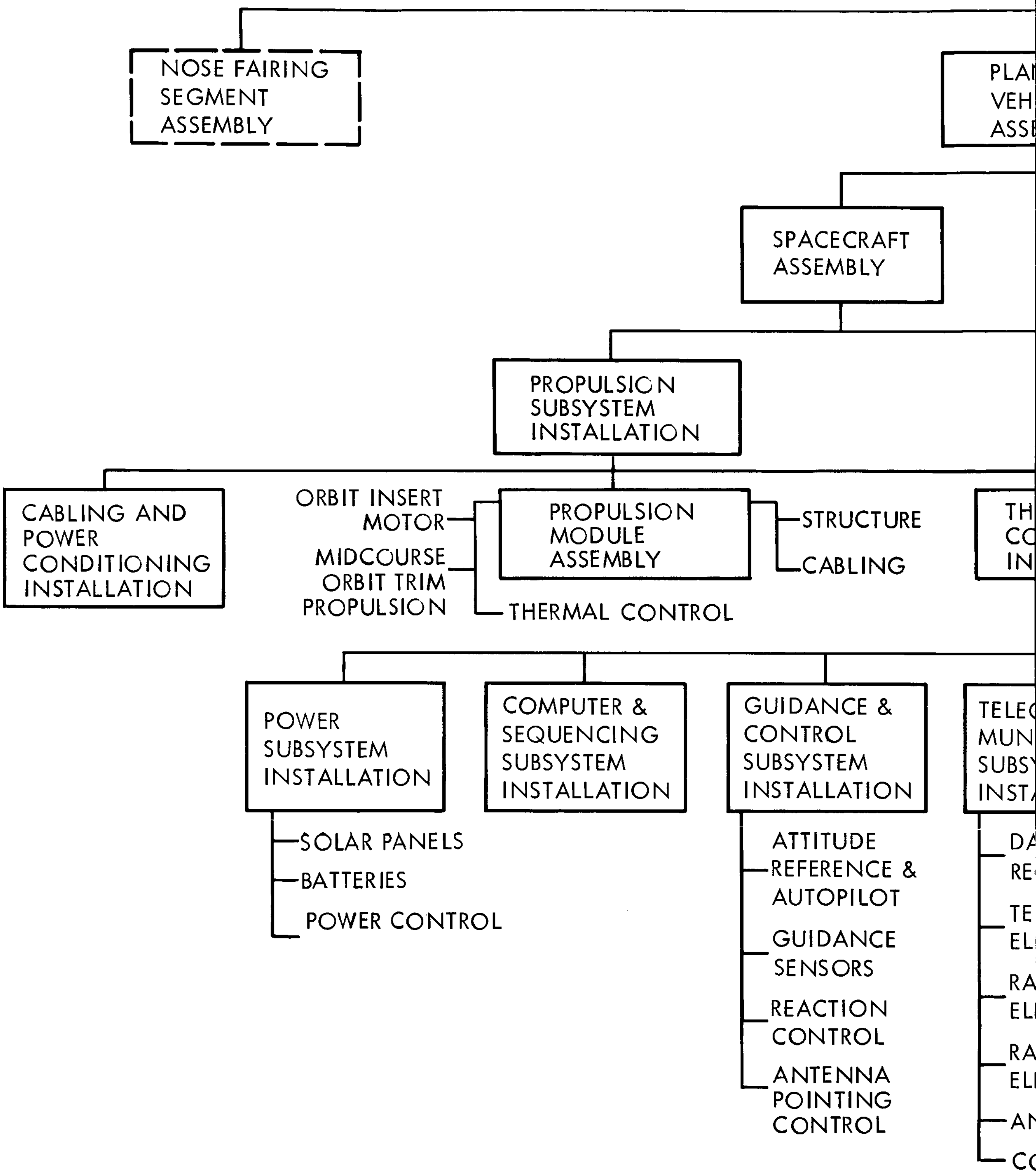
3.5 SYSTEM INTERFACE DEFINITION

This section describes the interfaces between the primary system elements of the Voyager Project, which are those defined by NASA/JPL in "Voyager 1971 Preliminary Mission Description," dated October 16, 1965. System elements and their respective interfaces are shown in Figure 3.5-1. Section references are included in the figure for location of specific interface details described elsewhere in Volumes A and B of this report. The details within the Spacecraft System block show the location of internal interface detail. Schedule interfaces affecting the Voyager Spacecraft System and its development are discussed in Section 5.3 of Volume A but are not referenced in the figure. The resulting preliminary interface definitions will be further defined during Phase IB.

3.5.1 Interface of Spacecraft System and Capsule System

The Spacecraft System (SCS) and Capsule System (CS) have physical, signal, power, and rf interfaces between the Flight Spacecraft and the Flight Capsule (Figure 3.5-2). The SCS and CS also have interfaces between their associated operational support equipment (OSE) in the system test complexes and in the launch complex equipment. Details of these interfaces are described in Section 3.1 of Volume B. The physical interfaces between the CS-OSE and the SCS-OSE will be the connectors on the interconnecting cables. An undefined interface--maintenance of the Flight

B L A N K


NOSE FAIRING SEGMENT ASSEMBLY
SPACECRAFT ASSEMBLY
PROPULSION SUBSYSTEM INSTALLATION
CABLING AND POWER CONDITIONING INSTALLATION
ORBIT INSERT MOTOR
MIDCOURSE ORBIT TRIM PROPULSION
PROPULSION MODULE ASSEMBLY
STRUCTURE
CABLING
THERMAL CONTROL
POWER SUBSYSTEM INSTALLATION
SOLAR PANELS
BATTERIES
POWER CONTROL
COMPUTER & SEQUENCING SUBSYSTEM INSTALLATION
GUIDANCE & CONTROL SUBSYSTEM INSTALLATION
ATTITUDE REFERENCE & AUTOPILOT
GUIDANCE SENSORS
REACTION CONTROL
ANTENNA POINTING CONTROL

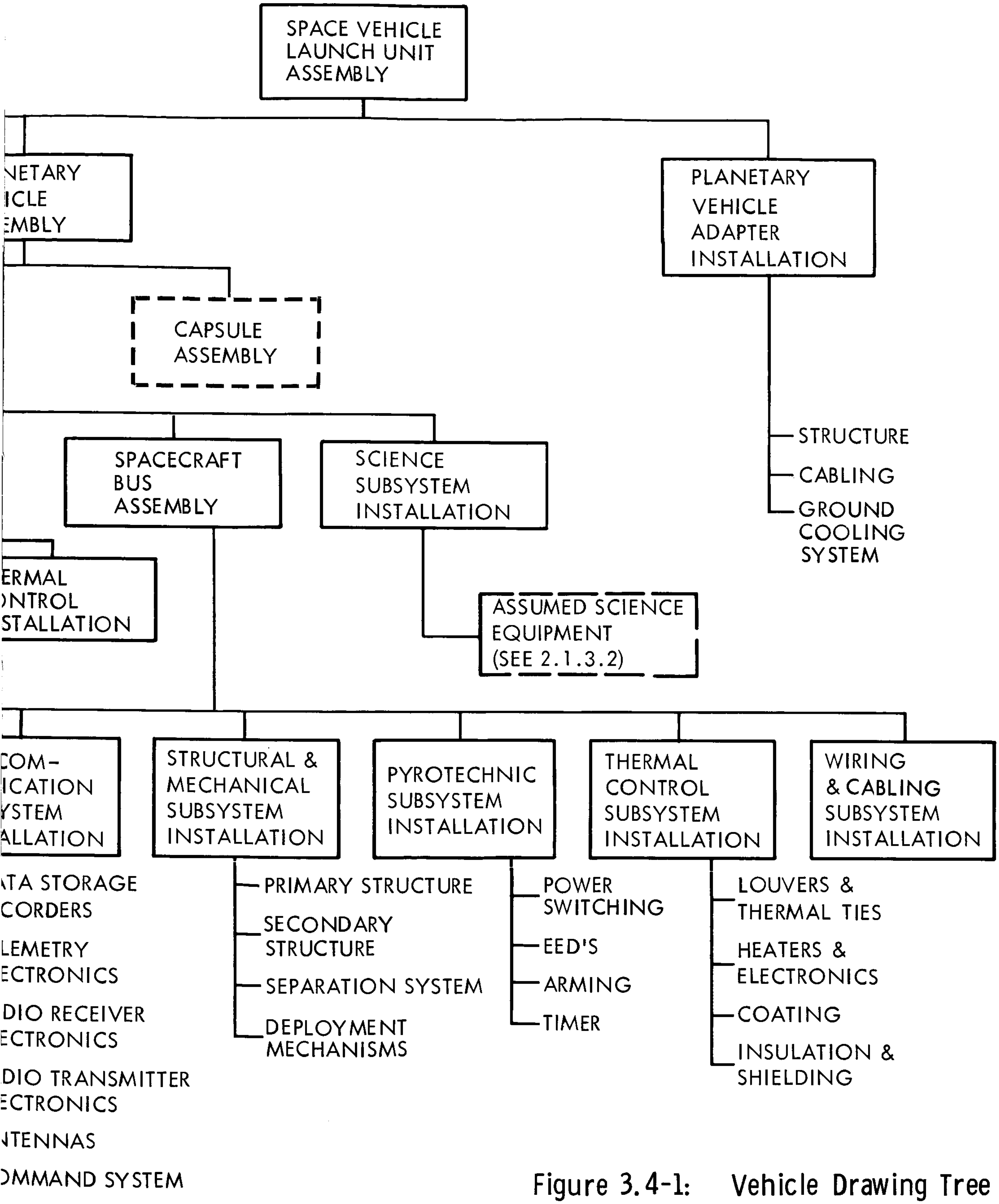
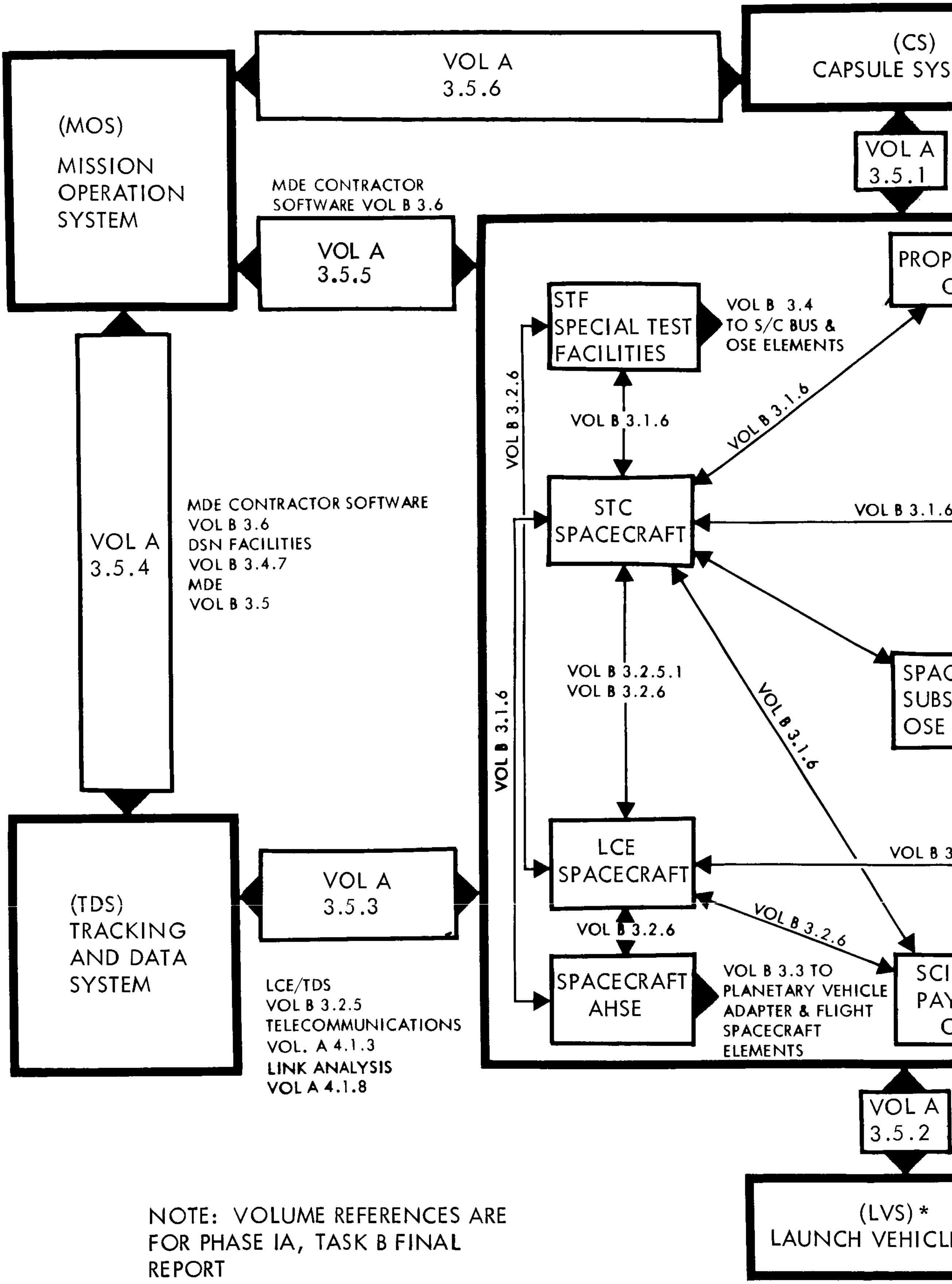
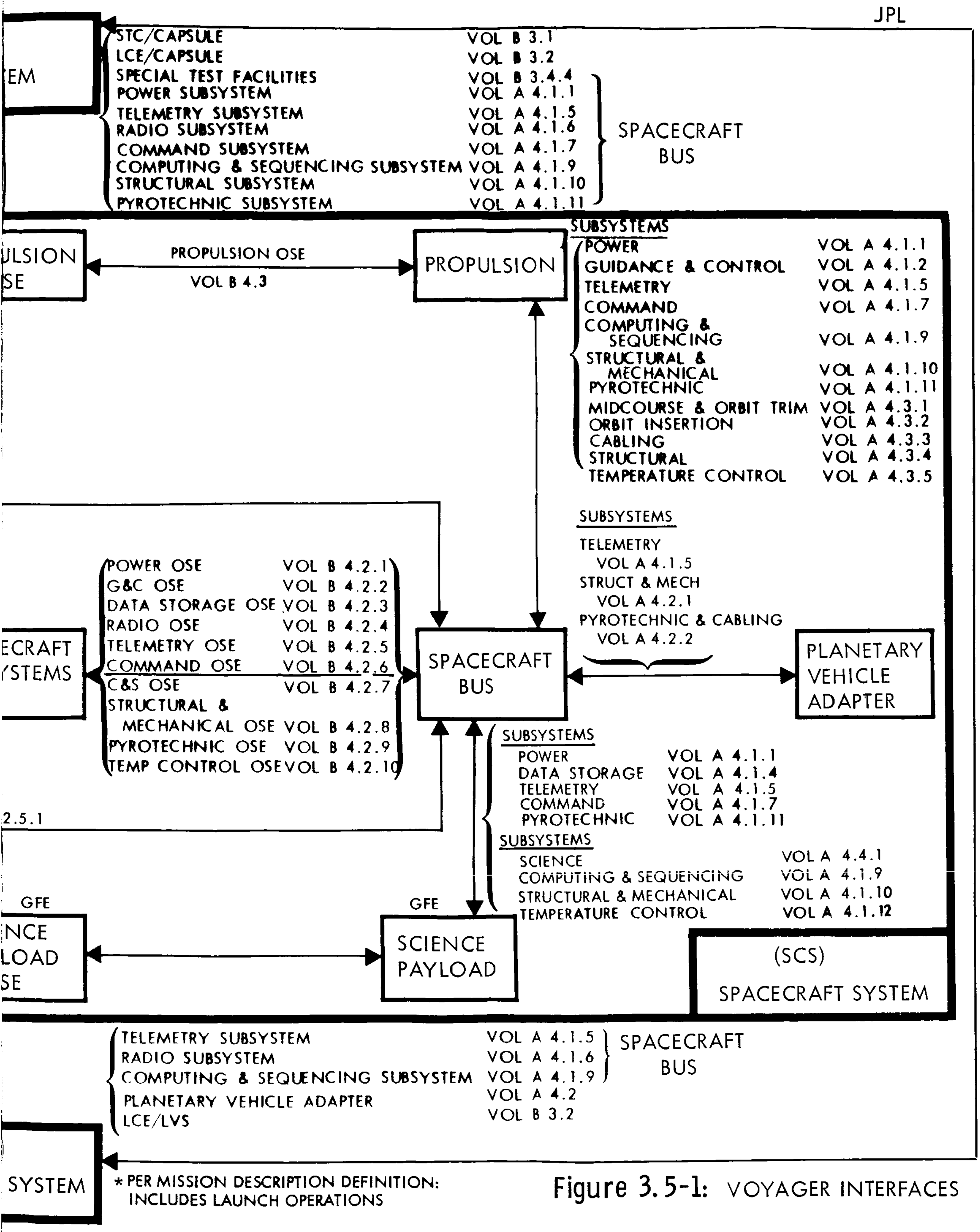


Figure 3.4-1: Vehicle Drawing Tree



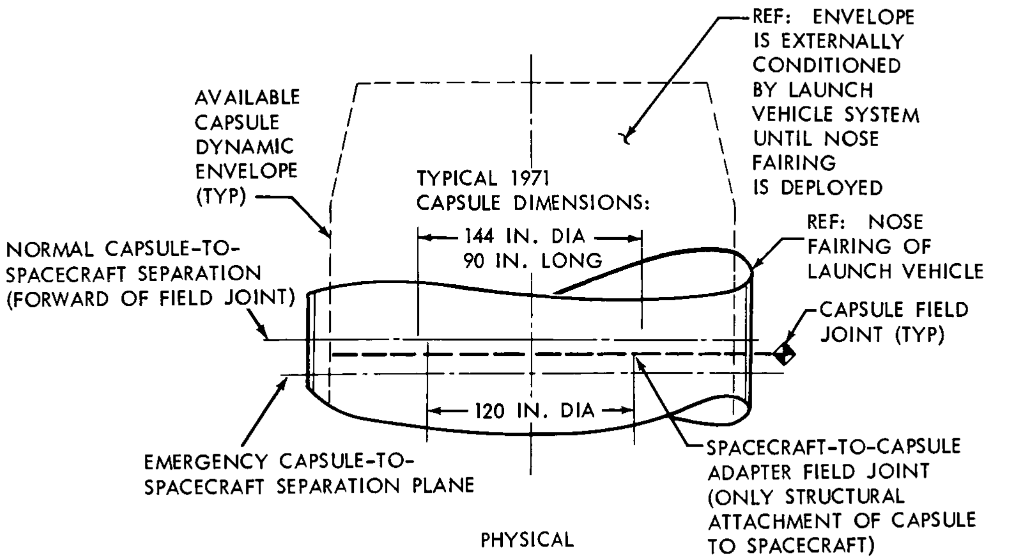
NOTE: VOLUME REFERENCES ARE FOR PHASE IA, TASK B FINAL REPORT



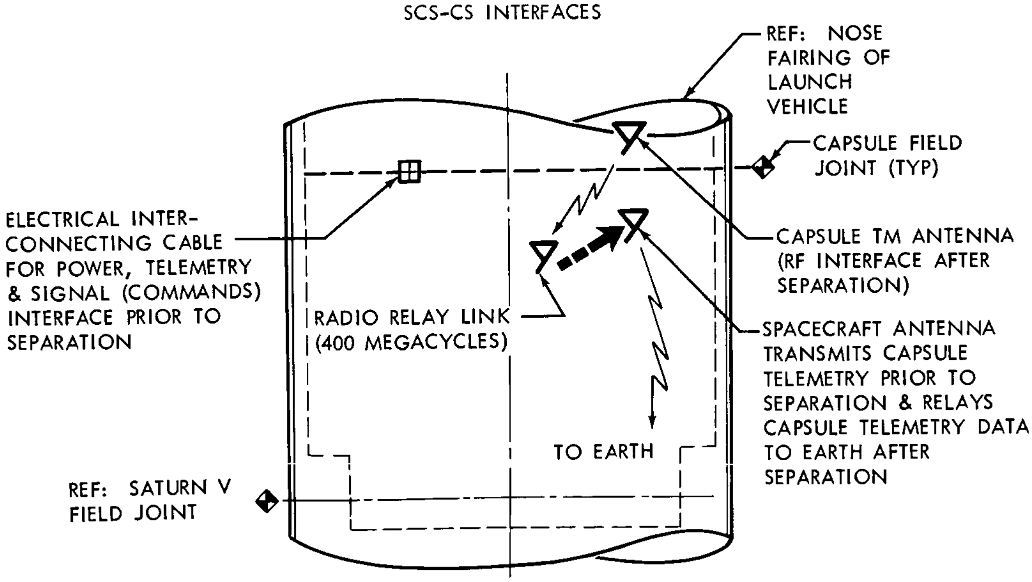
* PER MISSION DESCRIPTION DEFINITION:
INCLUDES LAUNCH OPERATIONS

Figure 3.5-1: VOYAGER INTERFACES

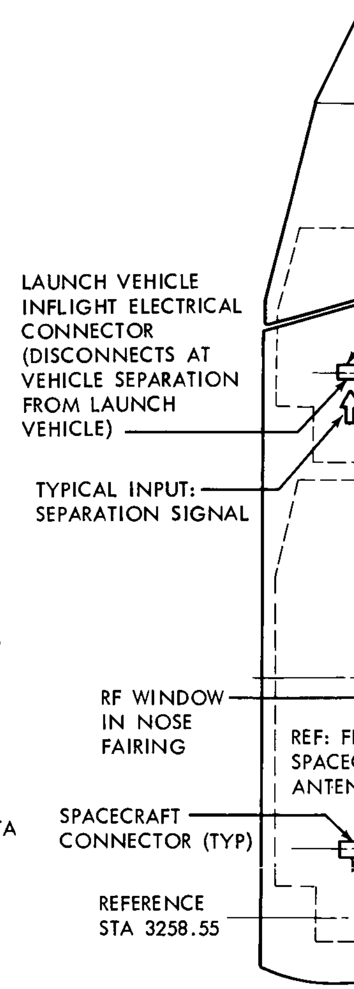
2



PHYSICAL
SCS-CS INTERFACES

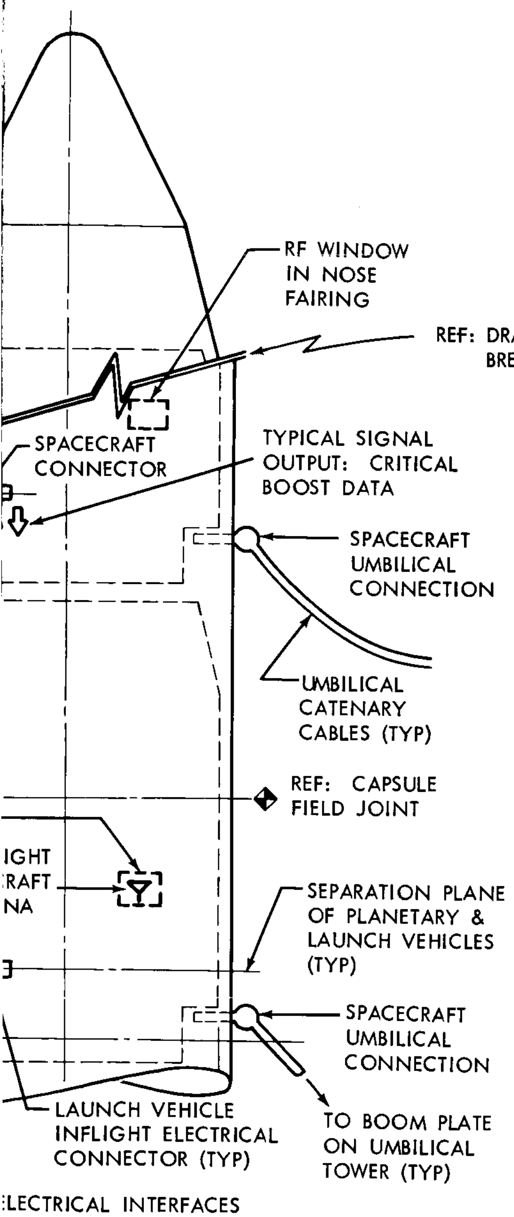


ELECTRICAL
SCS-CS INTERFACES

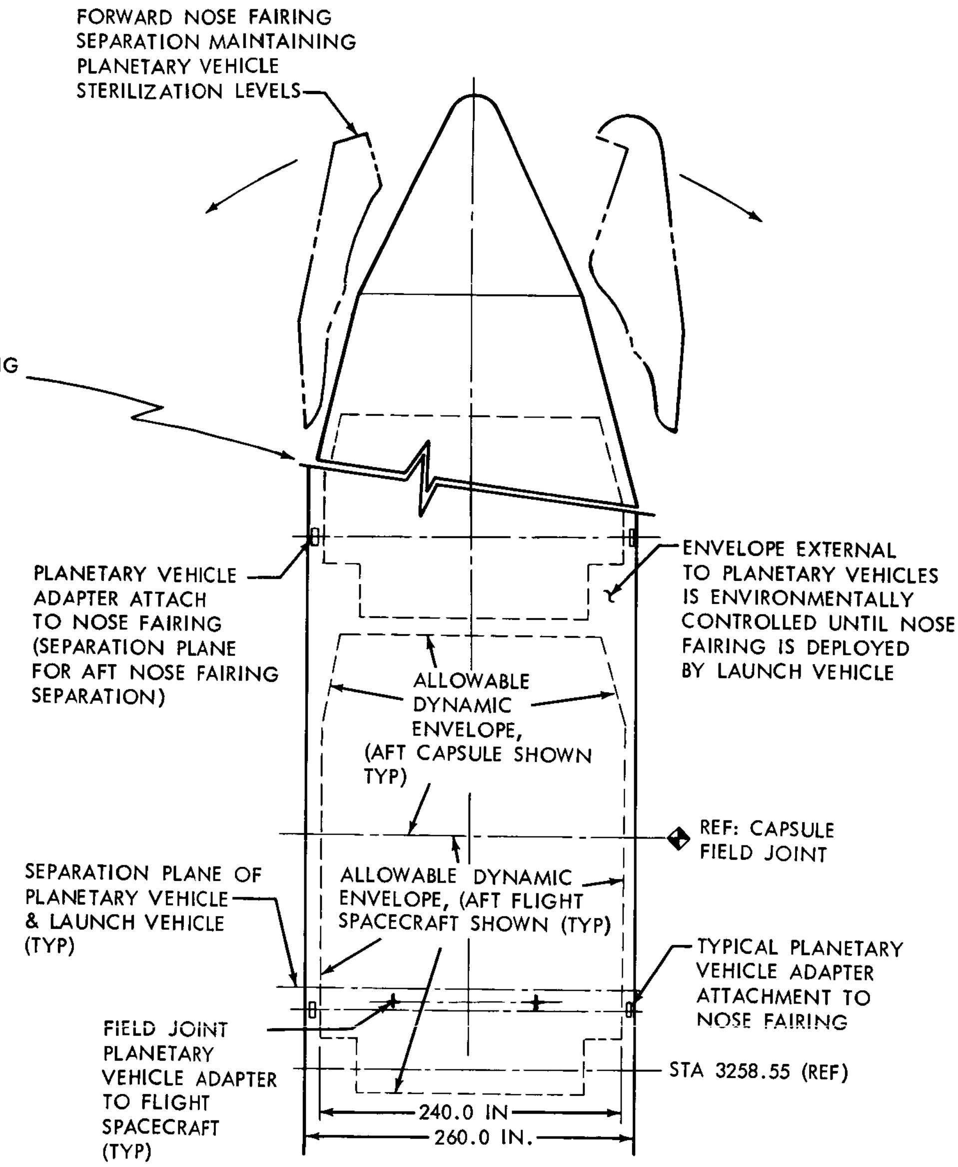


SCS-LVS

/



ELECTRICAL INTERFACES



SCS-LVS PHYSICAL INTERFACES

Figure 3.5-2: SCS/CS-SCS/LVS Interfaces

Capsule sterilization level--could cause problems. It will affect the interfaces of the Capsule System with both the SCS and the Launch Vehicle System.

3.5.2 Interface of Spacecraft System and Launch Vehicle System

The Spacecraft System (SCS) and the Launch Vehicle System (LVS) have physical, signal, power, rf, and environmental control interfaces between the Planetary Vehicle and the launch vehicle (Figure 3.5-2). Temperature, humidity, and cleanliness of environmental control under the nose fairing will be maintained by the LVS. Detailed definition of this interface will require coordination between the Spacecraft, Capsule, and Launch Vehicle Systems. The SCS and LVS have physical, power, air-conditioning, and signal interfaces between the SCS-LCE, the capsule LCE, and the launch complex. The physical interfaces will be the mechanical mounting fixtures for the LCE, the air-conditioning ducting terminations at the LCE, and the electrical connectors at the LCE. These interfaces are discussed in more detail in Section 3.2 of Volume B.

3.5.3 Interface of Spacecraft System and Tracking and Data System

The Spacecraft System (SCS) and Tracking and Data System (TDS) have command, telemetry, ranging, angle-tracking, and doppler-tracking interfaces with the Flight Spacecraft and all tracking and data stations assigned to the Voyager mission. These interfaces are contained in the S-band uplink and downlink. The uplink contains command, ranging, and doppler-tracking interfaces, whereas the downlink contains telemetry, ranging, and angle- and doppler-tracking interfaces. These links are discussed in Sections 4.1.3 and 4.1.8. The launch complex equipment (LCE)-TDS interfaces are covered in Section 3.2.5 of Volume B.

3.5.4 Interface of Tracking and Data System and Mission Operations System

The Tracking and Data System (TDS) and Mission Operations System (MOS) have signal, power, physical, programming, tracking-data-handling, telemetry, command-data-handling, and procedural interfaces. Mission-dependent hardware and software are necessary at DSIF's and the SFOF. For each signal path between these systems, there may be physical, signal-programming, procedural, and data-handling interfaces. Training, human engineering, and other TDS-MOS interfaces are described in Sections 3.4.7, 3.5, and 3.6 of Volume B.

3.5.5 Interfaces of Spacecraft System and Mission Operations System

The Spacecraft System and the Mission Operations System have functional interfaces in the command and telemetry data streams. Commands originated and verified by the mission operations teams of the SFOF constitute the command data stream, which crosses the MOS-TDS and TDS-SCS interfaces to the spacecraft. Data for command verification is fed back from the spacecraft to the MOS via the telemetry data stream. The telemetry data stream containing spacecraft and capsule data starts in the spacecraft and crosses the SCS-TDS and TDS-MOS interfaces before reaching the mission operations teams in the MOS. Possible interfaces exist where data from Spacecraft System tests and other Voyager Program system tests passes to the MOS. Training and procedural interfaces are discussed in Section 3.6 of Volume B.

3.5.6 Interface of Capsule System and Mission Operations System

The Capsule System (CS) and the Mission Operations System have a functional interface in the telemetry data stream. Telemetry data originating in the capsule crosses the SCS-CS interface to the spacecraft where it becomes a part of the telemetry data stream. Changes in mission requirements for later flights may affect these interfaces.

3.6 TELEMETRY CRITERIA

The following criteria are used to develop the preferred design of the telemetry subsystem and portions of associated telecommunication subsystem.

1) Provide data modes tailored to the requirements of each mission phase, but keep the total number of modes to a minimum consistent with the requirement to minimize equipment complexity. When requirements conflict, resolve the conflict in favor of maximizing the return of high-rate science data in the first month after Mars encounter.
2) Provide backup modes to transmit scientific and engineering data at reduced rates in degraded or emergency conditions.
3) Minimize antenna gain and, therefore, size, weight, and pointing accuracy requirements by using efficient coding and modulation techniques to the maximum extent possible without significantly increasing on-board complexity. Apply the weight saving to redundancy, and thereby increase probability of mission success.
4) Provide a balanced design that matches the data transmission capability to the total data acquisition and storage capability for at least the first 30 days after Mars encounter.

Applying the above criteria results in data storage, telemetry, and radio subsystems with the following characteristics and capabilities.

1) Two 1.2×10^8 bit recorders are provided for television data. In addition, after capsule impact and stored-capsule-data playback,

the 1.2 x 10^8 bit capsule recorder can be used for augmented storage of television data, providing a total picture storage capacity of 3.6 x 10^8 bits. This corresponds to data acquisition via the data automation system, at a rate of 50,000-bps for the 2-hour period on the sunlit side of Mars around the periapsis of a nominal 14-hour orbit. Lower capacity recorders for engineering and other scientific data increase the total data storage capacity to slightly over 3.8 x 10^8 bits.

2) The data rates provided for transmitting planetary science data are 7200 and 1200-bps. At the higher rate, 3.6 x 10^8 bits can be transmitted in approximately 14 hours. The entire recorder capacity can be filled and emptied during each 14-hour orbital period. Adding approximately 300-bps of engineering and low-rate science data provides a total data rate per spacecraft of 7500-bps. Thus, the two spacecraft from a single launch make full use of the maximum data handling capability of 15,000-bps at the deep-space stations for the time during which the 7200-bps high-rate data can be sustained by the spacecraft-to-ground link.

3) The following combination of telemetry and radio subsystem parameters maintains the 7200-bps rate (plus additional 288-bps on a separate subcarrier) for 80 days after encounter at a nominal encounter range of 160 x 10^6 km:

   a) Biorthogonal encoding (16, 5) of the serial data stream;

   b) Fifty-watt nominal rf power output;

   c) A 6.5-foot-diameter, doubly gimbaled, high-gain antenna.

   Although the biorthogonal encoding requires additional ground processing equipment, it reduces the high-gain-antenna diameter from 8 to 6.5 feet and adds only a few components to the telemetry subsystem. The reduced antenna size saves approximately 40 pounds in total spacecraft weight and relaxes pointing requirements. This system configuration will support 7488-bps rate for at least 80 days after encounter and 1260-bps (Mode 3) for an additional 100 days.

4) The Mariner IV fixed antenna is incorporated as a medium-gain backup to the high-gain antenna. It will support the 1260-bps rate for at least 100 days after encounter and 80-bps (Mode 1) for an additional 45 to 60 days.

A number of configuration variations are possible.

1) The third capsule 10^8 bit recorder can be held in reserve as a redundancy backup. A maximum data rate of 4800 bits per second will then transmit the entire storage capacity in one 14-hour orbit.
2) At 4800-bps, biorthogonal encoding can be eliminated or the high-gain antenna diameter can be reduced to 5 feet. Either change will still support 1200-bps at end of mission.
3) An additional mode (or modes) to provide a rate (or rates) intermediate between 7200 and 1200-bps can be added, with some increased complexity, to provide increased total data at end of mission.

These and other possible parameter variations will be explored with JPL during Phase IB.

A summary of proposed telemetry modes is presented in Tables 3.6-1 and 3.6-2.

Table 3.6-1: TELEMETRY MODULATION AS FUNCTION OF DATA MODE

MODE	DATA BIT-RATE (bits per second)		MODULATION	SUBCARRIER FREQUENCY (cps)		MISSION PHASE(S)
	Lower Channel	Upper Channel	Upper Channel / Lower Channel	Lower Channel	Upper Channel	
1	80		Coherent PSK/PM	1440	---	Launch Cruise
2	288	7200	Coded PSK/PM / Coherent PSK/PM	1440	92,160	Encounter Early Orbit
3	60	1200	Coded PSK/PM / Coherent PSK/PM	240	92,160	Late Orbit Prime, Early Orbit Backup
4	1.64		Two Channel Coherent PSK/PM	120 / 240	---	Emergency During Cruise and Orbit
5	--	---	Carrier Only	---	---	Acquisition

Table 3.6-2: AVERAGE DATA RATE (IN BITS) AS FUNCTION OF DATA MODE

MODE / DATA TYPE	1	2	3	4
	per sec.	per sec.	per sec.	per sec.
Spacecraft Engineering	12.3	44.3	11.5	1.45
Capsule Engineering	6.2	110.8	--	--
Cruise Science	36.9	44.3	30.0	--
Tape Recorder Playback*	15.4	55.4	11.5	--
Sync and Frame Ident.	9.2	33.2	6.9	0.19
TOTAL OF LOWER CHANNEL	80.0	288.0	60.0	1.64
Planetary Science (Upper Channel)	--	7200	1200	--

*Allocated to Cruise Science except during tape recorder playback.

3.7 TELEMETRY CHANNEL LIST

The Task B telemetry channel list is a complete revision of the list presented in Task A. This was necessary to conform to changes in the spacecraft subsystems due to the Saturn V innovation and changes in data transmission requirements and priority requirements established in "Performance and Design Requirements for the Voyager 1971 Spacecraft System, General Specification For--Preliminary," dated September 17, 1965. These requirements are in accord with the "Voyager 1971 Preliminary Mission Description," dated October 15, 1965.

The revision consists of assigning two channels for data transmission with five modes of operation to accommodate inflight selection of predetermined scientific, spacecraft engineering, and capsule data during the mission. Table 3.6-1 shows the characteristics of the five modes in respect to both channels. The types of data acquired in each mode with their average rates are shown in Table 3.6-2.

Table 3.7-1 summarizes spacecraft engineering measurements to be transmitted in real time during cruise; further details are available in Section 4.1.5. Similar telemetry-measurement lists for capsule engineering, cruise science, and planetary science will be generated when detailed science and capsule information becomes available.

3.8 GUIDANCE AND NAVIGATION MANEUVER ERROR ALLOCATIONS

The gross control accuracy requirements for velocity correction maneuvers were given in Section 2.3. This section presents the results of an analysis of the pointing and ΔV proportional and resolution accuracies obtained from the preferred design.

Table 3.7-1: Spacecraft Engineering Measurements

SPACECRAFT ENGINEERING MEASUREMENTS			SECONDS BETWEEN SAMPLES (MODE 1)					TYPES OF MEASUREMENTS: ELECTRICAL						DISCRETES					ANGLE			MISC.			DIGITAL				TM	
SUBSYSTEM	COMPONENT	TOTAL	10	60	120	600	1200	POWER	ANALOG VOLTAGE	EXCITATION VOLTAGE	CURRENT	PHASE ANGLE	SIGNAL LEVEL	SWITCH	ELECTRICAL THRESHOLD	EXCITATION	RPM SYNC	VEHICLE IDENTIFY	POINTING	POSITION	ATTITUDE ERROR	ATTITUDE RATE	PRESSURE	TEMPERATURE	PROGRAM DATA	COMMAND & IDENT.	FREQUENCY	EVENT COUNT	FRAMES SYNC	FRAME TIME
Power		40		5		18	17			12	14			3	1									10						
Computing & Sequencing		35		3			32								32										3					
Guidance & Control	Attitude Reference	20	2	2		10	6						2		4	2			8		3			1						
	Attitude Control	12		3		7	2									2	3				3	3		1						
	Autopilot	11				8	3		8	3																				
	Reaction Control	6				6																	2	4						
	Pointing Drivers	19					19							3									8	8						
Radio	Radio	33		4	3	4	22	8		6	4	2	4											9						
	Relay Radio	10				8	2		4				2		2									2						
Telemetry		32*		7*		23	2			8					11	8		1						2					1	1
Command		28		24	2		2								22											4	2			
Data Storage		42					42							14	14								7	7						
Structural & Mechanical		16					16							16																
Pyrotechnics		10				4	6							4		2												4		
Temperature Control		61					61													21				40						
Science		38					38		15															23						
Propulsion		12		5		7																	8	4						
	TOTALS	425*	2	53*	5	95	270	8	27	29	18	2	8	40	86	14	3	1	8	21	6	3	25	111	3	4	2	4	1	1
								92						144					35			139			13				2	

*includes frame time, sync, and vehicle identification

3.8.1 Pointing-Angle Accuracy

The contribution to pointing error of each significant instrument error was calculated. The error sources included Sun- and Canopus-sensor null offset, dead-zone and dead-zone switching-amplifier null offset errors, gyro drifts and misalignment, maneuver integrator calibration, resolution and drift errors, thrust-vector-control null errors, and center-of-gravity offset errors. In each case, the maximum value of the error source coefficient, as it appears in the error equation, was used. The G6B gyro performance characteristics are classified; therefore, G10B gyro numbers were used, which provided a conservative estimate. The results, given in Table 3.8-1, show that the pointing-accuracy requirement is met for all maneuvers.

3.8.2 ΔV Magnitude Accuracy

The ΔV magnitude accuracy for the midcourse and orbit-insertion maneuvers is given in Table 3.8-2. For both the midcourse corrections and delayed orbit insertion vernier corrections, ΔV magnitude accuracy is controlled by accelerometer measurement accuracy. Solid-motor orbit-insertion accuracy is governed by errors in the solid-motor impulse and the variation in secondary-injection thrust-control fuel used because of variation in thrust center-of-gravity misalignment.

3.9 FLIGHT SEQUENCE

This section describes the nominal sequence of operations performed by the spacecraft from the period immediately preceding launch until mission completion. Operational activities for the flight begin upon mission-acceptance review and the shipment of flight hardware to Kennedy

TABLE 3.8-1: POINTING ERROR ANALYSIS

ERROR SOURCE	UNITS	ERROR (3σ)	PITCH MULTIPLIER	MAX VALUE	YAW MULTIPLIER	MAX VALUE	SQUARED ERROR
MANEUVER ERROR							
Roll Drift Rate Breakdown	r/s						
Roll Gyro Uncompensated Bias	r/s	4.85×10^{-7}					
Roll Gyro Random Drift	r/s	7.28×10^{-8}					
Roll Gyro Bias, Temp Sensitive for 15°F	r/s	3.64×10^{-7}					
Roll Gyro Bias Stability	r/s	7.28×10^{-7}					
Roll Gyro Rate Integrator Drift	r/s	1.09×10^{-6}	$\left[\frac{1}{\dot\theta_y}(1-\cos\theta_y) - \sin\theta_y\left(\frac{\theta_R}{\dot\theta_R}+t_1\right)\right]$	571	0		7.2×10^{-7}
Roll Drift Rate RSS Total	r/s	1.448×10^{-6}					
Pitch Drift Rate	r/s	1.448×10^{-6}	$\left[-\frac{\sin\theta_y}{\dot\theta_y}+t_2\right]$	346	0		2.64×10^{-7}
Yaw Drift Rate	r/s	1.448×10^{-6}	0		$\left[\frac{\theta_y}{\dot\theta_y}+t_2\right]$	960	2.05×10^{-6}
Roll Gyro Misalignment with Pitch Axis	rad	8.93×10^{-4}	$-\sin\theta_R\cos\theta_y$	1	$(1-\cos\theta_R)$	2	3.99×10^{-6}
Roll Gyro Misalignment with Yaw Axis	rad	8.93×10^{-4}	$\cos\theta_y(1-\cos\theta_R)$	2	$\sin\theta_R$	1	3.99×10^{-6}
Pitch Gyro Misalignment with Roll Axis	rad	---	0	0	0	0	---
Pitch Gyro Misalignment with Yaw Axis	rad	8.93×10^{-4}	$\sin\theta_y$	1	0	0	7.99×10^{-7}
Yaw Gyro Misalignment with Pitch Axis	rad	8.93×10^{-4}	$\sin\theta_y$	1	0	0	7.99×10^{-7}
Yaw Gyro Misalignment with Roll Axis	rad	8.93×10^{-4}	$(1-\cos\theta_y)$	2	0	0	3.18×10^{-6}
Roll Limit Cycle Magnitude	rad	9.05×10^{-3}	$-\sin\theta_y$	1	0	0	81.5×10^{-6}
Roll Sensor Accuracy	rad	5.3×10^{-3}	$-\sin\theta_y$	1	0	0	28.1×10^{-6}
Roll Switching Amplifier Null Offset	rad	5.23×10^{-4}	$-\sin\theta_y$	1	0	0	2.73×10^{-7}
Pitch Limit Cycle Magnitude	rad	9.05×10^{-3}	$\cos\theta_y$	1	0	0	81.5×10^{-6}
Pitch Sensor Accuracy	rad	2.77×10^{-3}	$\cos\theta_y$	1	0	0	7.65×10^{-6}
Pitch Switching Amplifier Null Offset	rad	5.23×10^{-4}	$\cos\theta_y$	1	0	0	2.73×10^{-7}
Yaw Limit Cycle Magnitude	rad	9.05×10^{-3}	0	0	1	1	81.5×10^{-6}
Yaw Sensor Accuracy	rad	2.77×10^{-3}	0	0	1	1	7.65×10^{-6}
Yaw Switching Amplifier Null Offset	rad	5.23×10^{-4}	0	0	1	1	2.73×10^{-7}
Roll Turn-Time Resolution	sec	0.1	$-\dot\theta_R\ \sin\theta_y$	3.5×10^{-3}	0	0	1.225×10^{-8}
Roll Turn-Rate Calibration Error	r/s	1.15×10^{-6}	$\frac{\theta_R}{\dot\theta_R}\ \sin\theta_y$	225	0	0	6.71×10^{-8}
Roll Turn Torquer Current Error	rad	7.85×10^{-5}	$-\sin\theta_y$	1	0	0	6.16×10^{-11}
Yaw Turn-Time Resolution	sec	0.1	0	0	$\dot\theta_y$	3.5×10^{-3}	1.225×10^{-8}
Yaw Turn-Rate Calibration Error	r/s	1.15×10^{-6}	0		$\frac{\theta_y}{\dot\theta_y}$	900	1.07×10^{-6}
Yaw Turn Torquer Current Error	rad	3.15×10^{-4}	0	0	1	1	9.92×10^{-8}
						TOTAL	303.77×10^{-6}
					RSS MANEUVER ERROR =		17.42 m rad
THRUST VECTOR CONTROL ERRORS			PITCH ERROR		YAW ERROR		SQUARED ERROR
Midcourse	rad		2.69×10^{-3}		2.55×10^{-3}		13.74×10^{-6}
Orbit-Insertion Capsule Off	rad		7.31×10^{-3}		7.31×10^{-3}		106.8×10^{-6}
Orbit-Insertion Capsule On	rad		6.48×10^{-3}		6.48×10^{-3}		84.0×10^{-6}
Orbit-Trim Capsule Off	rad		9.28×10^{-3}		7.80×10^{-3}		147.0×10^{-6}
Orbit-Trim Capsule On	rad		2.48×10^{-3}		2.55×10^{-3}		12.65×10^{-6}

TOTAL POINTING ERRORS (3σ)	UNITS	CAPSULE ON	CAPSULE OFF
Midcourse	m rad	17.8 (1.02°)	
Orbit Insertion	m rad	19.7 (1.13°)	20.3 (1.16°)
Orbit Trim	m rad	17.8 (1.02°)	21.4 (1.23°)

Table 3.8-2: ΔV MAGNITUDE ERROR TABULATION

Midcourse and Orbit Insertion Vernier Corrections

Error Source	Magnitude Error, 3σ	ΔV Error, 3σ (Meters/Sec)
Accelerometer Null Bias	1×10^{-4} g	0.25% ΔV
Accelerometer Scale Factor	0.01%	0.01% ΔV
Accelerometer Resolution	0.012 m/sec	0.012
Engine Tail-Off Uncertainty	0.01 m/sec	0.01

Total rss Error, 3σ = $(0.0025\ \Delta V + 0.015)$ Meters/sec

Solid-Motor Orbit-Insertion Maneuver

Error Source	Magnitude Error, 3σ	ΔV Error 3σ (Meters/Sec)
Propellant Weight	0.15%	4.27
Propellant I_{sp}	0.45%	8.85
Inert Weight	10 lb	1.34
Midcourse Fuel Weight	10 lb	1.34
Secondary Injection Flow Rate	1 lb/sec	7.2

Total rss Error, 3σ = 12.3 Meters/sec

Space Center. Mission operations terminate at the end of Mars orbital operations.

The nominal flight sequence of operations for the 1971 Voyager represents an expansion of mission profile data provided by JPL. The nominal sequence for the Flight Spacecraft is summarized in Figure 3.9-1. The vertical lines on the correlation chart represent mission phases as contained in "Preliminary Voyager 1971 Mission Specification." Horizontal portions of the chart represent the Flight Spacecraft subsystems and their functions per flight phase. Any event or sequence of events within the described flight sequence can be altered by reprogramming the computing and sequencing (C&S) subsystem. The C&S subsystem can be reprogrammed any time prior to launch or while the spacecraft is in flight.

A discussion of the nominal sequence follows. Information additional to that contained in Figure 3.9-1 is presented for selected phases.

3.9.1 Prelaunch

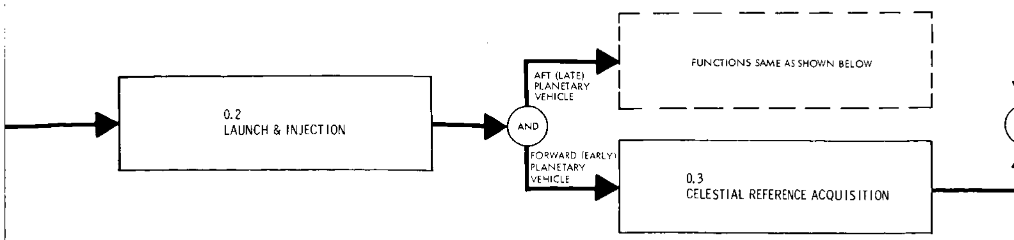
Prelaunch activities include final assembly, sterilization certification, system tests, and other activities resulting in the commitment to launch.

3.9.2 Launch and Injection (Figure 3.9-2)

Launch of the 1971 space vehicle takes place at ETR using the Saturn V three-stage launch vehicle. The Saturn V payload consists of two Planetary Vehicles and a nose fairing. Command carrier signals can be transmitted to the spacecraft via a low-gain antenna through the rf window in the nose fairing. RF signals from the payload are received by the launch-complex equipment (LCE) through the same antenna arrangement. The telemetry baseband is available on the umbilical and is analyzed by the LCE. Additional

0.1
PRELAUNCH

TIME		
	FORWARD PLANETARY VEHICLE	
	AFT PLANETARY VEHICLE	
LOS MOS DSN		A. MONITOR S/C CONDITIONED AIR B. CONDUCT S/C & LV CHECKS C. CHECK RANGE SAFETY D. RF & TELEMETRY CHECK E. EVALUATE TELEMETRY AT DSIF 71 F. VERIFY PV SYSTEMS BY RF LINK G. VERIFY PV OPERATION ON EXTERNAL POWER H. VERIFY PV OPERATION ON INTERNAL POWER I. PERFORM LCE SYSTEMS TESTS J. PERFORM AUTOMATIC COUNTDOWN K. COMMAND FIRST-STAGE IGNITION
POWER		A. SWITCH IN S/C INTERNAL POWER
COMPUTING & SEQUENCE (C&S)		A. START SPACECRAFT CLOCK (CUMULATIVE TIME REF) at T - 180 SEC
GUIDANCE AND CONTROL		
RADIO		A. RECEIVER A "ON" B. TRANSMIT SYSTEM STATUS TO LOS
TELEMETRY		A. SELECT MODE 1 B. ENGINEERING DATA (BOTH S/C & CAPSULE) RELAY TO LOS VIA HARDLINE
COMMAND		A. PROVIDE GROUND COMMANDS TO C&S
DATA STORAGE		
STRUCTURAL & MECHANICAL		A. PROVIDE STRUCTURAL SUPPORT TO S/C SUBSYSTEMS
PYROTECHNICS		
TEMPERATURE CONTROL		A. MONITOR FLIGHT SPACECRAFT TEMPERATURES
PROPULSION		
SCIENCE		A. ALL SCIENCE "OFF"
ADAPTER		
LAUNCH VEHICLE		A. PROVIDE S/C ENVIRONMENT CONTROL (PRECHILL TO $40^\circ \pm 5^\circ$ BY T - 1 HR) B. AUTOMATIC COUNTDOWN C. IGNITION S-I ENGINE



T = 0	+40 MIN ... 18 HR
T = 0	+55 MIN ... 18 H
RANGE A. MONITOR BOOSTER SEPARATION B. TRACK DURING BOOST C. RELAY TRACKING DATA TO LOS D. PROVIDE BOOSTER COMMANDS SFOF/MOS A. STANDBY ON ALERT B. ASSUME CONTROL AFTER INJECTION INTO HELIOCENTRIC ORBIT C. ANALYZE DATA & UPDATE TRAJECTORY PARAMETERS D. GENERATE & TRANSMIT DSIF TRACKING PREDICTION DATA DSIF 71 A. STANDBY ON ALERT B. RECEIVE ANTENNA SEARCH DATA C. SEARCH & ACQUIRE SPACE VEHICLE D. ESTABLISH & VERIFY CONTROL E. TRACK SPACECRAFT DSN A. RECEIVE DATA FROM DSIF 7 B. STANDBY C. TRACK	DSN A. MONITOR PV DOWNLINK TRANSMISSION B. ACHIEVE DOWNLINK/UPLINK PHASE LOCK C. MONITOR PV DEPLOYMENT • SOLAR PANELS • ANTENNAS • SCIENCE ELEMENTS D. RECEIVE SOLAR ACQUISITION DATA & RELAY E. RECEIVE & RELAY ROLL CALIBRATION DATA TO SFOF F. RECEIVE & RELAY CANOPUS ACQUISITION DATA TO SFOF G. RELAY BACKUP COMMAND REACQUIRE CANOPUS AS REQUIRED SFOF A. RECEIVE MONITOR DATA FROM DSN B. RECEIVE & ANALYZE ROLL CALIBRATION DATA C. RECEIVE & ANALYZE CANOPUS ACQUISITION DATA D. VERIFY CANOPUS ACQUISITION OR COMMAND CANOPUS REACQUISITION E. MONITOR PLANETARY VEHICLE • TRAJECTORIES & UPDATE • TRAJECTORY PARAMETERS
A. PROVIDE BATTERY POWER TO S/C SUBSYSTEMS	A. SENSE SOLAR PANEL OUTPUT B. SWITCH POWER SUBSYSTEM TO SOLAR POWER MODE C. TURN ON BATTERY CHARGER D. PROVIDE POWER TO S/C & FLIGHT CAPSULE
A. RECEIVE SEPARATION SIGNAL FROM MECHANISMS	A. COMMAND PYROTECHNICS UNLOCK & RELEASE ANTENNAS B. COMMAND PYROTECHNICS UNLOCK & RELEASE SCIENCE ELEMENTS MAGNETOMETER BOOM C. COMMAND PYROTECHNICS BACKUP COMMANDS (AS REQUIRED) D. COMMAND TELEMETRY MODE 1, MANEUVER "ON" E. COMMAND CRUISE SCIENCE "ON" F. COMMAND G&C TURN "ON" G&C EQUIPMENT G. COMMAND SOLAR ACQUISITION H. COMMAND COMMUNICATION PHASE LOCK I. COMMAND CALIBRATION ROLL J. COMMAND CELESTIAL REFERENCE "ON"
A. CANOPUS SENSOR PROTECTIVE SHUTTER "ON" B. G&C "OFF"	A. TURN ON G&C POWER B. GYROS IN RATE MODE, DAMP OUT PV TUMBLING RATE C. ENABLE SUN SENSOR D. PITCH & YAW PV, ACQUIRE SUN, SWITCH GYRO RATE "OUT" E. CALIBRATION ROLL F. STABILIZE PV DURING PHASE LOCK ACQUISITION • DERIVED RATE: PITCH & YAW AXES (SUN SENSOR) • SLEW ROLL AT .2°/SEC. G. CALIBRATE CANOPUS SENSOR MAGNETOMETERS & EARTH SENSOR H. RELAY STAR TRACKER & MAGNETOMETER DATA TO EARTH I. ACQUIRE CANOPUS • ENABLE CANOPUS SENSOR • SENSE CANOPUS PRESENCE + EARTH PRESENCE J. SWITCH TO CANOPUS REF., GYRO RATE "OUT" K. VERIFY CELESTIAL REF ACQUISITION (TO C&S) L. DISABLE CANOPUS HIGH INTENSITY GATE
A. TRANSMIT DOWNLINK SIGNAL B. TRANSMIT ENGRG. DATA	A. TRANSMIT DOWN LINK SIGNAL B. TRANSMIT VERIFICATION DATA — SOLAR ACQUISITION C. TRANSMIT ROLL CALIBRATION DATA D. TRANSMIT CANOPUS ACQUISITION DATA TO DSN E. RECEIVE GROUND BACKUP COMMANDS, IF REQUIRED
A. DATA RELAY TO BOOSTER TELEMETRY DURING POWERED FLIGHT B. RECEIVE & ENCODE ENGRG. DATA C. RECEIVE & ENCODE SEPARATION SIGNALS	A. TELEMETRY IN MODE 1 B. RECEIVE DEPLOYMENT VERIFICATION SIGNALS • SOLAR PANELS • SCIENCE ELEMENTS • ANTENNAS • GUIDANCE SCAN PLATFORM C. RECEIVE & ENCODE ROLL CALIBRATION DATA
	A. PROVIDE COMMAND BACKUP OF C&S AS REQUIRED B. RELAY GROUND COMMANDS TO C&S, IF REQUIRED
	A. STORE DATA DURING MANEUVER
A. PROVIDE STRUCTURAL SUPPORT TO S/C SUBSYSTEMS B. PROVIDE SEPARATION SENSOR C. PROVIDE SEPARATION SIGNALS TO TELEMETRY & C&S	A. DRIVE SOLAR PANELS TO LIMIT B. DRIVE ANTENNAS TO LIMIT C. DEPLOY GUIDANCE SCAN PLATFORM D. PROVIDE VERIFICATION SIGNALS TO TELEMETRY
A. ARM PYROTECHNICS	A. FIRE SOLAR PANEL SQUIBS
A. PROVIDE HEAT-SINK COOLING B. MONITOR FLIGHT S/C TEMPERATURES	A. PROVIDE HEAT-SINK COOLING B. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C. MONITOR FLIGHT S/C TEMPERATURES
	A. TURN ON CRUISE SCIENCE, GATHER DATA DURING CALIBRATION ROLL (DAE) B. RELAY ROLL CALIBRATION DATA TO TELEMETRY • REAL-TIME MAGNETOMETER DATA • MANEUVER MODE CRUISE DATA
A. PROVIDE RELEASE & SEPARATION MECHANISM FOR FWD & AFT PV'S B. PROVIDE SEPARATION SENSOR ACTIVATION	
A. LAUNCH: S-I & S-II BURN B. NOSE FAIRING JETTISON AT 350,000 FT. C. S-IVB FIRST BURN: INSERT IN PARKING ORBIT D. S-IVB SECOND BURN: INSERT IN TRANS-MARS TRAJECTORY E. PROVIDE SEPARATION SIGNAL TO FORWARD PV F. MAINTAIN ATTITUDE CONTROL DURING FORWARD PV SEPARATION G. FOLD BACK AFT NOSE FAIRING H. PROVIDE SIGNAL TO SEPARATE AFT PV I. MAINTAIN ATTITUDE CONTROL DURING AFT PV SEPARATION J. APPLY RETROTHRUST	

2

0.4 INTERPLANETARY CRUISE	0.5 INTERPLANETARY TRAJECTORY CORRECTION	0.6 FLIGHT CAPSULE CANISTER SEPARATION
6 MIN — 220 DAYS	1ST - 3.4 DAYS 2ND - 25 DAYS 3RD - 180 DAYS	210 DAYS — 220
21 MIN — 230 DAYS	1ST - 4.4 DAYS 2ND - 25 DAYS 3RD - 190 DAYS	220 DAYS — 230
DSN A. TRACK PV B. RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF C. PROVIDE PV RANGE DATA (DERIVED) D. COMMAND DATA PLAYBACK SFOF A. RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY B. RECEIVE & ANALYZE TRACK & RANGE DATA C. COMPUTE TRAJECTORY PARAMETERS	DSN A. TRANSMIT MANEUVER PARAMETERS TO PV • ROLL TURN MAGNITUDE & DIRECTION • YAW (OR PITCH) TURN MAGNITUDE & DIRECTION • MOTOR BURNTIME & ΔV B. TRANSMIT INITIATE MANEUVER SEQUENCE COMMAND C. RECEIVE VERIFICATION OF CORRECTION MANEUVER D. TRACK PV E. PROVIDE POINTING ANGLE & RANGE DATA TO ALL DSS F. RELAY DATA TO SFOF G. BACKUP CANOPUS ACQUISITION COMMAND SFOF A. COMPUTE MANEUVER PARAMETERS & TRANSMIT TO DSN B. VERIFY CORRECTION MANEUVER C. RECEIVE & DISPLAY ENGINEERING & SCIENCE DATA	DSN A. TRACK PV B. RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF C. PROVIDE PV RANGE DATA (DERIVED) D. COMMAND DATA PLAYBACK E. RECEIVE CANISTER SEPARATION VERIFICATION & RELAY TO SFOF F. TRANSMIT BACKUP COMMAND (AS REQUIRED) SFOF A. RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY B. RECEIVE & ANALYZE TRACK & RANGE DATA C. COMPUTE TRAJECTORY PARAMETERS D. RECEIVE VERIFICATION OF CANISTER SEPARATION
A. PROVIDE SOLAR POWER TO S/C SUBSYSTEMS B. PROVIDE POWER TO FLIGHT CAPSULE	A. PROVIDE SOLAR POWER TO S/C SUBSYSTEMS B. SWITCH TO BATTERY POWER C. NO POWER TO CAPSULE DURING BATTERY (OFF SUN) OPERATION D. SENSE SOLAR PANEL OUTPUT E. SWITCH POWER SUBSYSTEM TO SOLAR POWER MODE F. TURN ON BATTERY CHARGER G. PROVIDE POWER TO S/C & FLIGHT CAPSULE (CONTINUOUS)	A. PROVIDE SOLAR POWER TO ALL PV SUBSYSTEMS
A. COMMAND CANOPUS CONE ANGLE UPDATE B. COMMAND G&C SET HIGH-GAIN ANTENNA ELEVATION & AZIMUTH C. COMMAND RADIO SWITCH TO HIGH-GAIN ANTENNA (T + 80 TO 100 DAYS) D. COMMAND RADIO SWITCH TO HIGH-POWER TRANSMITTER (T + 3 DAYS) E. COMMAND RANGING "ON" & "OFF" F. COMMAND HIGH-GAIN ANTENNA POSITION UPDATE	A. RECEIVE MANEUVER PARAMETERS FROM COMMAND SUBSYSTEM B. COMMAND MANEUVER RECORDER "ON" COMMAND RADIO SWITCH TO LOW-GAIN ANTENNA C. COMMAND ACCELEROMETER "ON" D. COMMAND ATTITUDE CHANGE (ROLL, PITCH OR YAW) E. COMMAND AUTOPILOT OUTPUT SWITCH TO JET VANE ELECTRONICS F. COMMAND INERTIAL "HOLD" MODE FOR ΔV G. COMMAND PYROTECHNIC FIRE EED PRESSURIZE PROPELLANT (SQUIB VALVE) ▷1 H. COMMAND PYROTECHNIC FIRE EED, OPEN PROPELLANT VALVES ▷1 I. COMMAND CORRECTION MANEUVER FIRING J. COMMAND PYROTECHNIC FIRE EED, CLOSE PROPELLANT VALVES ▷2 K. COMMAND PYROTECHNIC FIRE EED, CLOSE PRESSURIZATION VALVE ▷2 L. COMMAND REACQUISITION OF CELESTIAL REF (PITCH OR YAW BACK, ROLL BACK) M. COMMAND SWITCH TO HIGH-GAIN ANTENNA (T + 80 DAYS) N. COMMAND MANEUVER RECORDER "OFF" SCIENCE IN CRUISE MODE O. COMMAND DATA STORAGE PLAYBACK STORED DATA TO DSN ▷1 1ST & 2ND CORRECTION ONLY---COMMAND SOLENOID VALVES ON SUBSEQUENT MANEUVERS ▷2 1ST CORRECTION ONLY, COMMAND PROPULSION SOLENOID VALVES ON SUBSEQUENT MANEUVERS	A. COMMAND MANEUVER RECORDER "ON" DATA RECORDING IN "MANEUVER" MODE B. COMMAND RADIO SWITCH TO LOW-GAIN ANTENNA C. COMMAND ATTITUDE CHANGE (ROLL, PITCH OR YAW) D. COMMAND PYROTECHNIC SUBSYSTEM FIRE CAPSULE CANISTER SEPARATION SQUIB E. COMMAND INERTIAL "HOLD" FOR TEN MINUTES F. COMMAND REACQUIRE CELESTIAL REFERENCES • (PITCH OR YAW BACK, ROLL BACK) G. COMMAND RADIO SWITCH TO HIGH-GAIN ANTENNA H. COMMAND MANEUVER RECORDER "OFF", CRUISE SCIENCE "ON" I. COMMAND DATA STORAGE PLAYBACK STORED DATA
A. UPDATE CANOPUS CONE ANGLE B. SET HIGH-GAIN ANTENNA ELEVATION & AZIMUTH C. ZERO GYRO INTEGRATORS, OBSERVE DRIFT, ADJUST GYRO BIAS D. REACQUIRE CELESTIAL REFERENCES (AS REQUIRED)	A. ZERO GYRO INTEGRATOR B. ROLL AND YAW (OR PITCH) C. SWITCH G&C OUTPUT TO JET VANE ELECTRONICS D. INERTIAL "HOLD" MODE FOR ΔV (CONTROL ATTITUDE WITH JET VANES) E. YAW BACK, REACQUIRE SUN F. ROLL BACK, REACQUIRE CANOPUS G. REMOVE GYRO DAMPING	A. ZERO GYRO INTEGRATOR B. ROLL, & PITCH OR YAW C. INERTIAL "HOLD" D. PITCH OR YAW BACK, ROLL BACK E. REMOVE GYRO DAMPING F. REACQUIRE CELESTIAL REFERENCES
A. TRANSMIT ENGINEERING DATA & SCIENCE DATA TO DSN B. TRANSMIT PLAYBACK DATA ON COMMAND (FROM DSN) C. SWITCH ON TWTA (AT 3 DAYS) D. RECEIVE COMMANDS FROM DSN E. MAINTAIN DOWNLINK PHASE LOCK	A. RECEIVE MANEUVER PARAMETERS FROM DSN & RELAY TO COMMAND SUBSYSTEM B. RECEIVE INITIATE MANEUVER COMMAND & RELAY TO COMMAND SUBSYSTEM C. SWITCH TO LOW-GAIN ANTENNA D. SWITCH TO HIGH-GAIN ANTENNA E. RELAY MANEUVER VERIFICATION TO DSN	A. TRANSMIT ENGINEERING & SCIENCE DATA TO DSN B. SWITCH TO LOW-GAIN ANTENNA C. SWITCH TO HIGH-GAIN ANTENNA D. REESTABLISH PHASE LOCK E. TRANSMIT PLAYBACK DATA ON COMMAND (FROM DSN) F. TRANSMIT CANISTER SEPARATION VERIFICATION TO DSN G. RECEIVE COMMANDS FROM DSN (AS REQUIRED)
A. ENCODE ENGINEERING & SCIENCE DATA - S/C B. RECEIVE FLIGHT CAPSULE STATUS INFO & ENCODE	A. MANEUVER RECORDER "ON" B. MANEUVER RECORDER "OFF"; PLAY BACK C. FORMAT DATA FOR TRANSMISSION (MODE 1)	A. ENCODE CANISTER SEPARATION VERIFICATION B. FORMAT DATA FOR TRANSMISSION (MODE 1)
A. FORMAT DATA FOR TRANSMISSION (MODE 1) B. PROVIDE C&S BACKUP AS REQUIRED C. RELAY GROUND COMMANDS TO C&S, IF REQUIRED	A. RECEIVE MANEUVER PARAMETERS FROM RADIO & RELAY TO C&S B. BACKUP C&S (AS REQUIRED)	A. PROVIDE DATA PLAYBACK COMMAND (AS REQD) B. PROVIDE C&S BACKUP AS REQD C. RELAY GROUND COMMANDS TO C&S AS REQUIRED
A. RECORD SCIENCE DATA (SOLAR FLARE MODE) B. PLAYBACK RECORDED DATA	A. RECORD MANEUVER DATA B. RELAY MANEUVER DATA TO RADIO (PLAY BACK)	A. RECORD MANEUVER DATA B. RELAY MANEUVER DATA TO TELEMETRY
A. PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS	A. PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS	A. PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS
	A. FIRE SQUIB, OPEN N_2 VALVE B. FIRE SQUIB, OPEN PROPELLANT VALVE } FIRST TWO MIDCOURSE MANEUVERS C. FIRE SQUIB, CLOSE PROPELLANT VALVE D. FIRE SQUIB, CLOSE N_2 VALVE } FIRST MANEUVER ONLY	A. FIRE EED, SEPARATE CAPSULE CANISTER
A. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING B. MONITOR FLIGHT SPACECRAFT TEMPERATURES	A. PROVIDE HEAT SINK COOLING B. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C. MONITOR FLIGHT S/C TEMPERATURE	A. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING B. MONITOR FLIGHT S/C TEMPERATURES
	A. OPEN N_2 VALVE, PRESSURIZE PROPELLANT B. OPEN PROPELLANT VALVE, PROVIDE THRUST C. PROVIDE THRUST VECTOR CONTROL (JET VANES) D. CLOSE PROPELLANT VALVE E. CLOSE N_2 VALVE	
A. GATHER CRUISE DATA B. SWITCH TO SOLAR FLARE MODE DURING SOLAR FLARE -SOLAR RADIATION & PARTICLE LEVEL THRESHOLD -TRAPPED RADIATION C. RETURN TO CRUISE MODE AFTER FLARE	A. GATHER CRUISE DATA	A. GATHER CRUISE DATA

0.7
MARS ORBIT INSERTION

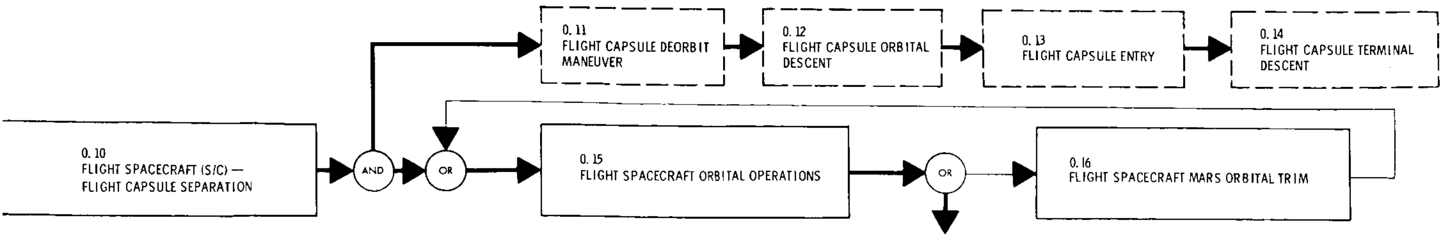
OR

0.8
PLANETARY VEHICLE (PV) ORBITAL OPERATIONS

OR

0.9
PLANETARY VEHICLE MARS ORBIT TRIM

DAYS	220 DAYS	22
DAYS	230 DAYS	23
DSN A. RELAY ORBIT INSERTION PARAMETERS B. TRACK PV C. RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF D. PROVIDE PV RANGE DATA (DERIVED) E. COMMAND DATA PLAYBACK F. RECEIVE VERIFICATION OF ORBIT INSERTION SEQUENCE & RELAY TO SFOF SFOF A. COMPUTE ORBIT INSERTION PARAMETERS & PROVIDE BACKUP COMMANDS AS REQD B. RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY C. RECEIVE & ANALYZE TRACK & RANGE DATA D. COMPUTE TRAJECTORY PARAMETERS E. RECEIVE ORBIT INSERTION VERIFICATION	DSN A. TRACK PV B. RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF C. PROVIDE PV RANGE DATA (DERIVED) D. COMMAND DATA PLAYBACK E. RECEIVE MARS ORBIT DATA; RELAY TO SFOF SFOF A. RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY B. RECEIVE & ANALYZE TRACK & RANGE DATA C. COMPUTE TRAJECTORY PARAMETERS D. RECEIVE & DISPLAY MARS ORBIT DATA E. COMPUTE ORBIT PARAMETERS	DSN A. TRANSMIT MANEUVER PARAMETERS TO PV • ROLL TURN MAGNITUDE AND DIRECTION • YAW (OR PITCH) TURN MAGNITUDE & DIRECTION • MOTOR BURNTIME & ΔV B. TRANSMIT INITIATE MANEUVER SEQUENCE COMMAND C. RECEIVE VERIFICATION OF CORRECTION MANEUVER D. TRACK PV E. PROVIDE POINTING ANGLE & RANGE DATA TO ALL DSS SFOF A. COMPUTE MANEUVER PARAMETERS & TRANSMIT TO DSN B. VERIFY CORRECTION MANEUVER C. RECEIVE & DISPLAY ENGINEERING & SCIENCE DATA
A. PROVIDE SOLAR POWER TO S/C SUBSYSTEMS B. SWITCH TO BATTERY POWER C. SENSE SOLAR PANEL OUTPUT D. SWITCH POWER SUBSYSTEM TO SOLAR POWER MODE E. TURN ON BATTERY CHARGER F. PROVIDE POWER TO S/C & FLIGHT CAPSULE	A. PROVIDE SOLAR POWER TO S/C SUBSYSTEMS B. PROVIDE POWER TO FLIGHT CAPSULE C. SWITCH TO BATTERY POWER DURING SOLAR OCCULTATION ▷ ▷ SUN OCCULTATION CONTINGENT UPON ORBIT SELECTION	A. PROVIDE SOLAR POWER TO S/C SUBSYSTEMS B. SWITCH TO BATTERY POWER C. NO POWER TO CAPSULE DURING BATTERY (OFF SUN) OPERATION D. SENSE SOLAR PANEL OUTPUT E. SWITCH POWER SUBSYSTEM TO SOLAR POWER MODE F. TURN ON BATTERY CHARGER G. PROVIDE POWER TO S/C & FLIGHT CAPSULE (CONTINUOUS)
A. COMMAND PLANET SCIENCE "ON" (AFTER TWO ORBITS) B. COMMAND G&C PERFORM PREMANEUVER SEQUENCE C. COMMAND G&C PERFORM ORBIT INSERTION MANEUVER D. COMMAND PYROTECHNICS IGNITE ORBIT INSERTION ENGINE E. RECEIVE MANEUVER VERIFICATION FROM G&C F. COMMAND TELEMETER ENCODE MANEUVER VERIFICATION TO RADIO & TRANSMIT	A. COMMAND LIMB/TERMINATOR UNIT "ON" B. COMMAND ORBITAL SCIENCE "ON"; PLANET SCIENCE "OFF", CRUISE "OFF" C. COMMAND CANOPUS ANGLE UPDATE D. COMMAND HIGH-GAIN ANTENNA POSITION UPDATE E. COMMAND DEPLOY SCIENCE SCAN PLATFORM F. COMMAND RANGING "ON" & "OFF"	A. RECEIVE MANEUVER PARAMETERS FROM COMMAND SUBSYSTEM B. COMMAND MANEUVER RECORDER "ON" C. COMMAND SELECT LOW-GAIN ANTENNA D. COMMAND ACCELEROMETER "ON" E. COMMAND ATTITUDE CHANGE (ROLL, & PITCH OR YAW) F. COMMAND AUTOPILOT OUTPUT SWITCH TO JET VANE ELECTRONICS G. COMMAND INERTIAL HOLD MODE FOR ΔV H. COMMAND PROPULSION SYSTEM ACTIVATE I. COMMAND CORRECTION MANEUVER FIRING J. COMMAND TERMINATE THRUST K. COMMAND TVC "OFF" L. COMMAND PROPULSION SYSTEM DEACTIVATE M. COMMAND REACQUISITION OF CELESTIAL REF (PITCH OR YAW BACK, ROLL BACK) N. COMMAND MANEUVER RECORDER "OFF" SCIENCE IN ORBIT MODE O. COMMAND SELECT HIGH-GAIN ANTENNA P. COMMAND TELEMETRY & RADIO TRANSMIT STORED DATA TO DSN
A. SENSE PLANET APPROACH GEOMETRY, RELAY TO TELEMETRY B. ZERO GYRO INTEGRATOR C. ROLL, & PITCH OR YAW D. INERTIAL HOLD MODE FOR ΔV E. POINT HIGH-GAIN ANTENNA TO EARTH F. ENABLE INSERTION ENGINE THRUST VECTOR CONTROL (TVC) AUTOPILOT OUTPUTS TO TVC G. YAW BACK TO PREVIOUS ATTITUDE H. ACQUIRE SUN, SWITCH GYRO RATE "OUT" (PITCH & YAW AXES) I. ROLL BACK TO PREVIOUS ATTITUDE J. ACQUIRE CANOPUS, SWITCH GYRO RATE "OUT" ROLL AXIS K. CANOPUS ACQUISITION GATE SIGNAL TO C&S L. POINT HIGH-GAIN ANTENNA TO EARTH	A. SENSE CANOPUS OCCULTED; SWITCH ROLL AXIS TO INERTIAL "HOLD" B. REACQUIRE CANOPUS; SWITCH ROLL AXIS IRU "OUT" C. SENSE LIMB & TERMINATOR CROSSINGS, RELAY TO TELEMETRY & G&C D. UPDATE CANOPUS CONE ANGLE E. UPDATE HIGH-GAIN ANTENNA POSITION (AS REQD) F. SENSE SUN OCCULTED; SWITCH PITCH & YAW AXES TO INERTIAL "HOLD" ▷ G. SENSE SUN PRESENCE, SWITCH GYRO RATE "OUT" ▷ ▷ SUN OCCULTATION DURING THIS PHASE OF THE MISSION IS CONTINGENT UPON THE ORBIT SELECTED	A. ZERO GYRO INTEGRATORS B. ROLL, YAW (OR PITCH) C. SWITCH G&C OUTPUT TO JET VANE ELECTRONICS D. INERTIAL "HOLD" MODE FOR ΔV E. YAW BACK; REACQUIRE SUN F. ROLL BACK; REACQUIRE CANOPUS G. REMOVE GYRO DAMPING
A. RECEIVE INSERTION PARAMETERS FROM DSN (AS REQUIRED) & RELAY TO C&S B. SWITCH TO LOW-GAIN ANTENNA C. SWITCH TO HIGH-GAIN ANTENNA D. REESTABLISH PHASE LOCK E. TRANSMIT ENGINEERING DATA & SCIENCE DATA TO DSN F. TRANSMIT ORBIT INSERTION VERIFICATION	A. MAINTAIN DOWNLINK PHASE LOCK B. RECEIVE DSN COMMANDS C. TRANSMIT ENGINEERING DATA TO DSN (MARS ACQUISITION & PV EPHEMERIS) D. EARTH OCCULTATION - REESTABLISH PHASE LOCK	A. RECEIVE MANEUVER PARAMETERS FROM DSN & RELAY TO COMMAND SUBSYSTEM B. RECEIVE INITIATE MANEUVER COMMAND & RELAY TO COMMAND SUBSYSTEM C. SWITCH TO LOW-GAIN ANTENNA D. SWITCH TO HIGH-GAIN ANTENNA E. RELAY MANEUVER VERIFICATION & DATA TO DSN
A. FORMAT DATA FOR TRANSMISSION (MODE 1) • ENGINEERING & SCIENCE DATA • FLIGHT CAPSULE STATUS INFORMATION AND ENCODE	A. SCAN PLATFORM DEPLOYMENT VERIFICATION B. FORMAT DATA FOR TRANSMISSION (MODE 2)	A. TURN "ON" MANEUVER RECORDER B. TURN "OFF" MANEUVER RECORDER & PLAY BACK C. FORMAT DATA FOR TRANSMISSION (MODE 1)
A. BACKUP C&S AS REQUIRED B. RECEIVE UPDATED ORBIT PARAMETERS FROM DSN (AS REQUIRED)	A. BACK UP C&S AS REQUIRED B. RELAY GROUND COMMANDS TO C&S AS REQUIRED	A. RECEIVE MANEUVER PARAMETERS FROM RADIO, RELAY TO C&S B. BACKUP C&S AS REQUIRED
A. RECORD DATA DURING MANEUVER B. PLAY BACK RECORDED DATA	A. RECORD ENGINEERING & SCIENCE DATA (MODE 2) B. PLAY BACK RECORDED DATA	A. RECORD MANEUVER DATA B. RELAY MANEUVER DATA TO TELEMETRY
A. PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS	A. PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS B. DEPLOY SCIENCE SCAN PLATFORM C. VERIFICATION SIGNAL TO TELEM	A. PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS
A. FIRE EED's; ENERGIZE TVC SYSTEM B. FIRE EED's; IGNITE SOLID INSERTION MOTOR		
A. PROVIDE HEAT SINK COOLING B. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C. MONITOR FLIGHT S/C TEMPERATURE	A. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING B. MONITOR FLIGHT S/C TEMPERATURES	A. PROVIDE HEAT-SINK COOLING B. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C. MONITOR FLIGHT S/C TEMPERATURE
A. PRESSURIZE TVC FREON TANK B. THRUST VECTOR CONTROL "ON"; INPUT TO G&C IRU & ACCELEROMETER C. IGNITE PROPELLANT—ORBIT INSERTION ENGINE		A. OPEN N_2 VALVE, PRESSURIZE PROPELLANT B. OPEN PROPELLANT VALVE, PROVIDE THRUST C. CLOSE PROPELLANT VALVE D. CLOSE N_2 VALVE
A. GATHER CRUISE DATA	A. TURN OFF CRUISE SCIENCE B. TURN ON ORBITAL SCIENCE ▷ ▷ OPTIONAL; ENGINEERING DATA & MANEUVER DURING EARLY ORBITAL OPERATIONS MAY REQUIRE SCIENCE TO BE SWITCHED OFF	A. GATHER CRUISE SCIENCE DATA DURING MANEUVER
		FLIGHT CAPSULE



	225 DAYS	405 DAYS — 227 DAYS
	236 DAYS	405 DAYS — 239 DAYS
:K PV IVE ENGINEERING & SCIENCE \ & RELAY TO SFOF /IDE RANGE DATA IVE SEPARATION DATA FICATION Y BACKUP COMMANDS FROM AS REQUIRED SFOF A. RECEIVE SCIENCE/ENGINEERING DATA AND DISPLAY B. RECEIVE & ANALYZE TRACK & RANGE DATA C. RECEIVE CAPSULE SEPARATION VERIFICATION D. GENERATE BACKUP COMMANDS & RELAY TO DSN (INCLUDING CAPSULE SEPARATION COMMAND)	DSN A. TRACK S/C B. RECEIVE ENGINEERING & SCIENCE DATA C. RECEIVE CAPSULE DATA D. PROVIDE RANGE & TRACK DATA E. COMMAND DATA PLAYBACK SFOF A. RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY B. RECEIVE & ANALYZE TRACK & RANGE DATA	DSN A. TRANSMIT MANEUVER PARAMETERS TO S/C • ROLL TURN MAGNITUDE & DIRECTION • YAW (OR PITCH) TURN MAGNITUDE & DIRECTION • MOTOR BURNTIME & ΔV B. TRANSMIT INITIATE MANEUVER SEQUENCE COMMAND C. RECEIVE VERIFICATION OF CORRECTION MANEUVER D. TRACK S/C E. PROVIDE POINTING ANGLE & RANGE DATA TO ALL DSS SFOF A. COMPUTE MANEUVER PARAMETERS & TRANSMIT TO DSN B. VERIFY CORRECTION MANEUVER C. RECEIVE & DISPLAY ENGINEERING & SCIENCE DATA
/IDE SOLAR POWER TO S/C & FLIGHT ULE CH CAPSULE POWER "OFF" CH TO BATTERY POWER CH POWER SUBSYSTEM BACK TO SOLAR PANEL MODE; CHARGE BATTERIES	A. PROVIDE SOLAR POWER TO S/C B. PROVIDE BATTERY POWER DURING OCCULTATION C. SENSE SOLAR PANEL OUTPUT, SWITCH TO SOLAR POWER D. CHARGE BATTERY	A. PROVIDE SOLAR POWER TO S/C SUBSYSTEMS B. SWITCH TO BATTERY POWER C. SENSE SOLAR PANEL OUTPUT D. SWITCH POWER SYSTEM TO SOLAR POWER MODE E. TURN ON BATTERY CHARGER
IMAND RADIO TURN RELAY RECEIVERS ON IMAND DATA STORAGE TO MODE 2 NSFER DATA TO CAPSULE (AS REQUIRED) MMAND CAPSULE SWITCH TO INTERNAL POWER MMAND CAPSULE RECORDER "ON" MMAND PREMANEUVER SEQUENCE MMAND ATTITUDE CHANGE (ROLL, AND PITCH OR YAW) MMAND PYROTECHNICS FIRE EED, SEPARATE CAPSULE UMBILICAL MMAND PYROTECHNICS FIRE EED, SEPARATE CAPSULE MMAND REACQUISITION OF CELESTIAL REFERENCES CH OR YAW BACK, ROLL BACK)	A. COMMAND CAPSULE RECORDER PLAYBACK B. COMMAND RELAY RECEIVER OFF AFTER CAPSULE IMPACT C. SWITCH TO DATA MODE 2 RECORD DATA DURING EARTH OCCULTATION D. RECEIVE LIMB & TERMINATOR CROSSINGS, COMMAND SCIENCE SUBSYSTEM "ON" & "OFF" AS DESIRED E. COMMAND DATA TRANSFER TO DAE (AS REQUIRED) F. COMMAND CANOPUS CONE ANGLE UPDATE G. COMMAND HIGH-GAIN ANTENNA POSITION UPDATE	A. RECEIVE MANEUVER PARAMETERS FROM COMMAND SUBSYSTEM B. COMMAND MANEUVER RECORDER "ON"; SCIENCE IN MANEUVER MODE C. COMMAND PREMANEUVER SEQUENCE D. COMMAND ACCELEROMETER "ON" E. COMMAND SWITCH TO LOW-GAIN ANTENNA F. COMMAND ATTITUDE CHANGE • ROLL • PITCH OR YAW G. COMMAND AUTOPILOT OUTPUT SWITCH TO JET VANE ELECTRONICS H. COMMAND INERTIAL "HOLD" MODE FOR ΔV I. COMMAND PROPULSION SYSTEM ACTIVATE J. COMMAND CORRECTION MANEUVER FIRING K. COMMAND TERMINATE THRUST L. COMMAND JET VANE TVC "OFF" M. COMMAND PROPULSION SYSTEM DEACTIVATE N. COMMAND REACQUISITION OF CELESTIAL REF • PITCH OR YAW BACK • ROLL BACK O. COMMAND MANEUVER RECORDER "OFF" SCIENCE IN ORBIT MODE P. COMMAND SWITCH TO HIGH-GAIN ANTENNA Q. COMMAND DATA STORAGE PLAYBACK TO DSN
GYRO INTEGRATOR , YAW (OR PITCH) TIAL "HOLD" MODE FOR CAPSULE SEPARATION ILIZE S/C BACK, ROLL BACK QUIRE CELESTIAL REFERENCES, SWITCH GYRO RATE "OUT"	A. SENSE SUN OCCULTATION; SWITCH TO INERTIAL "HOLD" PITCH & YAW AXES B. SENSE SOLAR PRESENCE; SWITCH GYRO RATE "OUT" PITCH & YAW AXES C. SENSE CANOPUS OCCULTATION; SWITCH TO INERTIAL "HOLD" MODE, ROLL AXIS D. REACQUIRE CANOPUS; SWITCH GYRO RATE OUT; ROLL AXIS E. SENSE LIMB & TERMINATOR CROSSINGS, RELAY TO C&S F. UPDATE CANOPUS CONE ANGLE G. UPDATE HIGH-GAIN ANTENNA POSITION (AS REQUIRED)	A. ZERO GYRO INTEGRATORS B. ROLL, PITCH (OR YAW) C. SWITCH G&C OUTPUT TO JET VANE ELECTRONICS D. INERTIAL "HOLD" MODE FOR ΔV E. PITCH OR YAW BACK; REACQUIRE SUN F. ROLL BACK; REACQUIRE CANOPUS G. REMOVE GYRO DAMPING
Y RECEIVER "ON" Y CAPSULE VERIFICATION TO S/C PRIOR TO SEPARATION IVE CAPSULE DATA Y DATA TO CAPSULE RECORDER	A. RECEIVE CAPSULE DATA & RELAY TO CAPSULE DATA RECORDER B. RELAY RECEIVER "OFF" C. TRANSMIT CAPSULE DATA PLAYBACK TO DSN D. EARTH OCCULTATION - REESTABLISH PHASE LOCK	A. RECEIVE MANEUVER PARAMETERS FROM DSN & RELAY TO COMMAND SUBSYSTEM B. RECEIVE INITIATE MANEUVER COMMAND & RELAY TO COMMAND SUBSYSTEM C. SWITCH TO LOW-GAIN ANTENNA D. SWITCH TO HIGH-GAIN ANTENNA E. RELAY MANEUVER VERIFICATION TO DSN
MAT DATA FOR TRANSMISSION (MODE 2) C DATA IGHT CAPSULE DATA	A. FORMAT DATA FOR TRANSMISSION (MODE 2)	A. TURN "ON" MANEUVER RECORDER B. TURN "OFF" MANEUVER RECORDER & PLAY BACK C. FORMAT DATA FOR TRANSMISSION (MODE 2)
UP C&S AS REQUIRED GROUND COMMANDS TO C&S AS REQUIRED	A. BACK UP C&S AS REQD B. RELAY GROUND COMMANDS TO C&S AS REQUIRED	A. RECEIVE MANEUVER PARAMETERS FROM RADIO; RELAY TO C&S B. RELAY GROUND COMMANDS TO C&S AS REQUIRED C. PROVIDE BACKUP TO C&S (AS REQUIRED)
CH TO MODE 2 /IDE RECORDED DATA SEQUENCE TO CAPSULE	A. SWITCH TO RECORD MODE 2 B. RECORD DATA DURING EARTH OCCULTATION	A. RECORD MANEUVER DATA B. RELAY MANEUVER DATA TO RADIO (PLAY BACK)
IDE STRUCTURAL SUPPORT TO S/C SYSTEMS	A. PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS	A. PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS
EED'S, SEPARATE CAPSULE UMBILICAL EED'S, SEPARATE CAPSULE		
IDE HEAT-SINK COOLING IDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING ITOR FLIGHT S/C TEMPERATURE	A. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING B. MONITOR FLIGHT S/C TEMPERATURES	A. PROVIDE HEAT-SINK COOLING B. PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C. MONITOR FLIGHT S/C TEMPERATURE
		A. OPEN N_2 VALVE, PRESSURIZE PROPELLANT B. OPEN PROPELLANT VALVE, PROVIDE THRUST C. CLOSE PROPELLANT VALVE D. CLOSE N_2 VALVE
SE SCIENCE "ON" ON CAPSULE DATA RECORDER	A. RECORD CAPSULE DATA B. PLAY BACK CAPSULE DATA C. ORBIT SCIENCE "ON" & "OFF" PER C&S COMMANDS	A. ORBITAL SCIENCE "ON" B. SWITCH TO MANEUVER MODE (CRUISE SCIENCE "ON")
ON INTERNAL POWER LISH RADIO LINK WITH S/C VE DATA; UPDATE CAPSULE PROGRAM (AS REQUIRED) RATE FROM S/C OST SMIT DATA	A. TRANSMIT DATA	

Figure 3.9-1: Nominal Flight Sequence

5

0.2 LAUNCH & INJECTION

NO.	EVENT	SOURCE *	DESTINATION *	TIME (T-1560 MIN. / T = 0 / T + 40 MIN. / T + 55 MIN.)
1.	VERIFY AND CONTINUOUS MONITOR OF S/C AIR CONDITIONING	S/C	LCE	T-1560 MIN.
2.	LOAD S-IV HYPERGOLICS (REF.)			T-1550
3.	START S/C SYSTEM CHECKS, START SCIENCE SUB-SYSTEM AUTOCALIBRATION, START CAPSULE CHECKS	LCE	P/V	T-1480
4.	CONDUCT RF AND T/M CHECKS	S/C	LOS	T-1250
5.	PERFORM POWER TRANSFER TEST			T-855
6.	LV PREPARATION & SAFETY CHECKS			T-255
7.	START FINAL S/C CHECKS	LCE	S/C	T-90
8.	CONDUCT POWER TRANSFER TEST	LCE	S/C	T-45
9.	START LV PNEUMATIC SYSTEM PRESSURIZATION (REF.)			T-187 SEC.
10.	START AUTOMATIC COUNTDOWN			T-187 SEC.
11.	COMMAND 1ST STAGE IGNITION (REF.)			T-7 SEC.
12.	LIFT OFF			T-0
13.	BURN S-IC STAGE			T + 153 SEC.
14.	SEPARATE S-IC STAGE			T + 147 SEC.
15.	BURN S-II STAGE			T + 522 SEC.
16.	SEPARATE FORWARD PORTION OF NOSE FAIRING AND JETTISON (ALTITUDE OF 350,000 ft.)			
17.	SEPARATE S-II STAGE			T + 522 SEC.
18.	BURN S-IV STAGE (ORBIT INSERTION)			T + 524 SEC — T + 20 MIN., 34 SEC.
19.	PARKING ORBIT COAST (2 TO 90 MINUTES)			2 TO 90 MINUTES DURATION 2
20.	INTERPLANETARY TRANSIT TRAJECTORY INJECTION (REF.)			1 T + 39 MIN.
21.	PLANETARY VEHICLE SEPARATION			
	a. SEPARATE FORWARD PLANETARY VEHICLE	SIVB	FWD. PV	T + 40 MIN. (1 MIN.)
	b. FOLD BACK AFT PORTION OF NOSE FAIRING	LV		
	c. SEPARATE REAR PLANETARY VEHICLE	SIVB	AFT PV	T + 55 MIN. (15 MIN.)
22.	S-IVB RETROFIRE (REF)			T + 115 MIN. (60 MIN.)

FORWARD PV SEPARATION

AFT PV SEPARATION

TIME IN MINUTES UNLESS OTHERWISE NOTED.

1 NOMINAL SECOND BURN - 5 MIN., 15 SEC.

2 NOMINAL

* FUNCTIONS WITHOUT SOURCE, DESTINATION ARE LAUNCH VEHICLE CONTRACTOR RESPONSIBILITY, AND ARE SHOWN HERE FOR CONTINUITY PURPOSES.

Figure 3.9-2: Flight Sequence — Launch and Injection

Planetary Vehicle functions are monitored over umbilical hardlines during countdown. The launch vehicle provides environmental control for the payload.

From liftoff, through boost, injection, and subsequent celestial reference acquisition, the radio subsystem operates in the standard receive mode and a low-power transmit mode. The launch transmitter operates over a low-gain antenna to transmit telemetry information and give initial acquisition capability prior to utilization of the high-power traveling-wave tube amplifiers (TWTA's). Mode 1 (as defined in Table 3.6-1) is the primary data mode used during all flight phases prior to Mars encounter. This mode transmits cruise-science, capsule, and spacecraft-engineering data. The Cape Kennedy station provides telemetry reception from liftoff to local horizon. Telemetry data is also transmitted by the S-IVB telemetry system during boost and parking-orbit phases.

The forward nose fairing, which includes everything back to the forward Planetary Vehicle Adapter, is jettisoned at 350,000 feet during S-IIB burn. The S-IVB first burn injects the S-IVB and payload into an Earth-parking orbit.

After 2 to 90 minutes in the parking orbit, the launch vehicle injects the payload on a trans-Mars trajectory and provides the signal to initiate separation of the forward Planetary Vehicle. The launch vehicle then provides the signal to fold back the rear Planetary Vehicle fairing elements; 15 minutes later it provides the signal to separate the rear Planetary Vehicle. Sixty minutes after aft Planetary Vehicle separation, the S-IVB stage retrothrusts to deflect the S-IVB/rear nose fairing assembly from the trans-Mars trajectory.

3.9.3 Celestial Reference Acquisition (Figure 3.9-3)

The automatic celestial-reference acquisition procedure is as follows: C&S command turns on gyros and switches power to autopilot. The gyros are in rate mode and initial vehicle rates are damped. Power during boost and acquisition is supplied by batteries. The solar panels, antennas, magnetometer boom, and the guidance scan platform are unlatched and deployed. Pitch-and-yaw attitude-error signals from the Sun sensors cause the autopilot to drive the reaction jets, rotating the vehicle to reduce the error. With no position signal, the roll axis remains in rate-damping mode. Solar acquisition is nominally completed within 30 minutes after separation.

When Sun acquisition is accomplished and the roll axis is pointing toward the Sun, a signal from a Sun-gate causes a constant rate command to be applied to the roll autopilot input. The resulting maneuver allows calibration of the magnetometer and Canopus and Earth sensors. The outputs of these sensors are transmitted to the ground station. At the end of the time allotted for the calibration maneuver (about 16 hours from launch for Mariner) the automatic Canopus-acquisition sequence is initiated. Roll continues at constant rate in the direction that, according to the predicted star pattern, ensures the greatest probability of acquiring Canopus. Onboard logic is activated which combines the outputs of the Canopus sensor intensity gates and the Earth sensor.

When an acquirable object enters the Canopus sensor field of view, and the Earth sensor is also illuminated, the onboard logic initiates the following actions: The roll-slew command is replaced by the star-tracked error signal as the controlling input to the roll autopilot and, after

0.3 CELESTIAL REFERENCE ACQUISITION

FLIGHT SEQUENCE

NO.	EVENT	SOURCE	DESTINATION
1.	G&C Power On	C&S	G&C
2.	Provide Spacecraft Battery Power	Power	S/C
3.	Acquire Engineering Data	Telemetry	Radio
4.	Provide Downlink Transmission	Radio	DSN
5.	Provide Spacecraft Environmental Control	Temp	S/C
6.	Deploy Solar Panels		
	a. Fire Squibs	SIT	Pyrotech
	b. Back-up Deployment Command	C&S	Pyrotech
	c. Drive Panels to Limit Stops	Mech	
7.	Deploy Antennas		
	a. Deploy Low-Gain Antennas		
	1. Unlock & Release	C&S	Pyrotech
	2. Backup Deploy Signal (if required)	CMD	Pyrotech
	3. Move Antenna to Operation Position & Lock	Mech	
	b. Deploy Medium-Gain Antennas		
	1. Unlock & Release Signals	C&S	Pyrotech
	2. Backup Release Signal (if required)	CMD	Pyrotech
	3. Move Antennas to Operating Positions	Mech	
	c. Deploy High-Gain Antennas		
	1. Unlock & Release Signal	C&S	Pyrotech
	2. Backup Unlock & Release Signal (if required)	CMD	Pyrotech
	3. Move Antenna to Operating Position and Lock	Mech	
8.	Deploy Science Elements		
	a. Unlock & Release Magnetometer Boom	C&S	Pyrotech
	b. Backup Deployment Signal	CMD	Pyrotech
9.	Deploy Guidance Scan Platform		
	a. Unlock and Release Guidance Scan Platform	C&S	Pyrotech
	b. Backup deployment signal	CMD	Pyrotech
10.	Turn on Cruise Science	C&S	Science
11.	Provide Attitude Stabilize Command	C&S	G&C
12.	Reduce P/V Tumbling Rate from Separation	Auto Pilot	Reaction Control
13.	Acquire Sun	Auto Pilot	Reac. Control
	a. Provide Acquire Celestial References Command	C&S	G&C
	b. Pitch and Yaw S/C to Orient Roll Axis Toward Sun	Reaction Control	

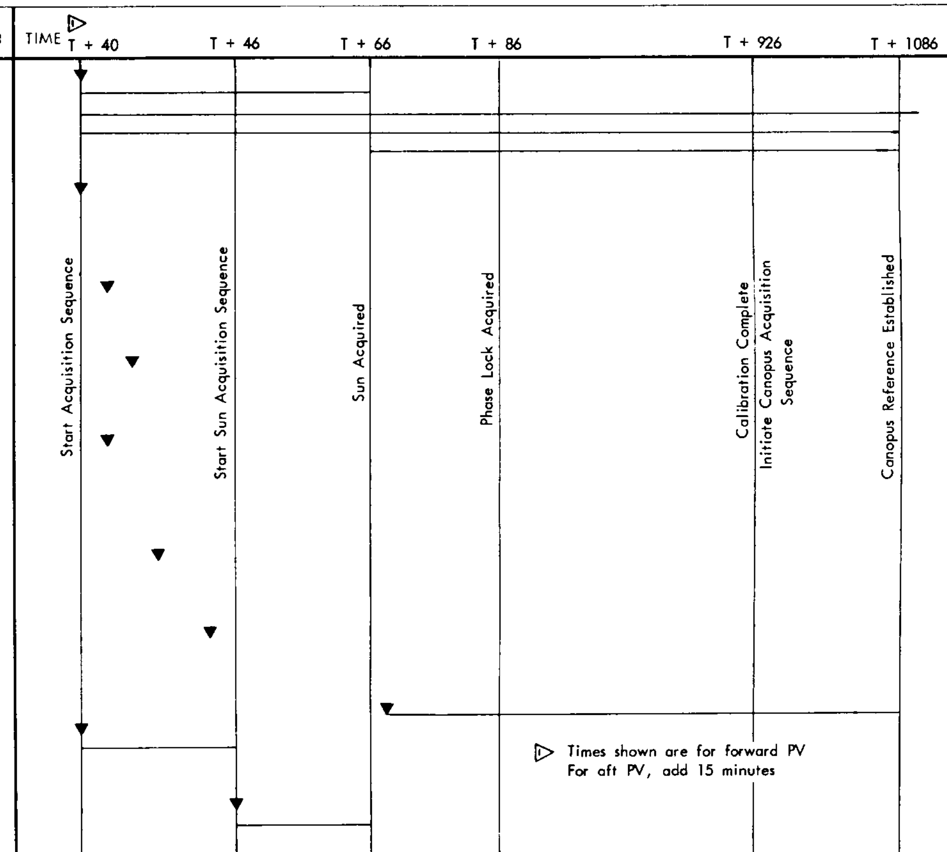


Figure 3.9-3: Flight Sequence — Celestial Reference

0.3 CELESTIAL REFERENCE ACQUISITION (Con't.) — FLIGHT SEQUENCE

NO.	EVENT	SOURCE	DESTINATION
14.	Provide Spacecraft Solar Power	Power	All S/S
15.	Switch Batteries to Battery Charger	Power	—
16.	Provide Power to Flight Capsule	Power	Capsule
17.	Shield Flight Capsule from Sun	S/C	—
18.	Provide Uplink Transmission	DSN	Radio
19.	Establish Phase Lock		
	a. Downlink Phase Lock	Radio	DSN
	b. Uplink Phase Lock	DSN	Radio
20.	Maintain Sun Reference	Sun Sensor	Reaction Control
21.	Maintain Inertial Reference — Roll Axis Only	G&C	Reaction Contr. Control
22.	Perform Calibration Roll		
	a. Turn on Caopus Sensor, Earth Sensor and Select Canopus Cone Angle	C&S	G&C
	b. Provide Calibration Commands	C&S	Science
	c. Initiate Roll	C&S	G&C
	d. Calibrate Earth Sensor	G&C	TM/Radio
	e. Calibrate Magnetometer	Science	TM/Radio
	f. Antenna Signal Strength		DSN
	g. Terminate Roll Signal	C&S	G&C
23.	Acquire Canopus		
	a. Provide Celestial Reference Command	C&S	G&C
	b. Enable Canopus Acquisition Gate		
	c. Roll until Simultaneous Canopus Sensor/Earth Sensor Outputs Detected		
	d. Canopus Sensor Error Signal to Roll Autopilot		
	e. Remove Roll Slew ———		
	f. Hold Period for Ground Override Command		
	g. Switch to Canopus Reference		
24.	Process Data (Mode 1)	F/C, Science, S/C	Telemetry
25.	Transmit Data to Earth (Mode 1)	Radio	DSN
26.	Ground Verification of Canopus Acquisition	DSN/SFOF	
27.	Ground Override Command — Reacquire Canopus (If required)	DSN	Radio

▷ Times shown are for forward PV for aft PV, add 15 minutes

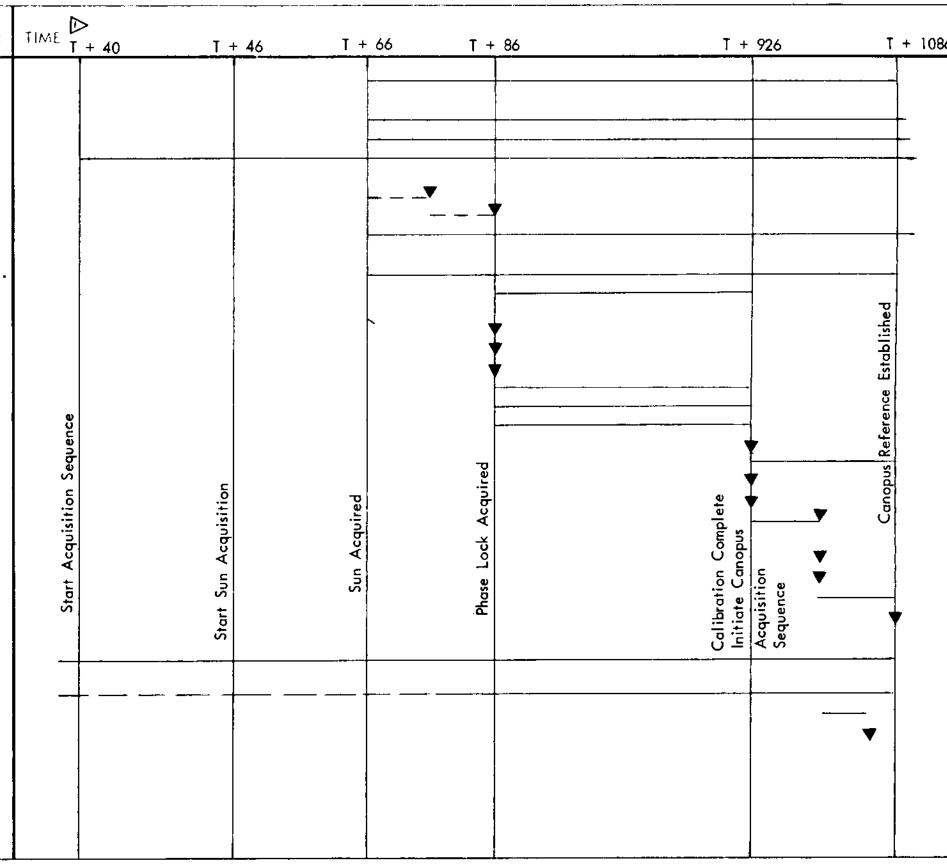


Figure 3.9-3: (Continued) Flight Sequence — Celestial Reference

a time delay of several hours to allow for ground evaluation, the derived-rate compensation is switched in to replace the gyros as a source of rate information for damping. The Earth-presence condition on Canopus acquisition is removed later.

The ground facility will monitor and analyze all information developed during the automatic procedure. The ground facility can override the automatic sequence since it has the following flight-generated data: magnetometer calibration information, star-intensity map (Canopus sensor calibration), Earth-sensor intensity map, and antenna-signal-strength data, all correlated to the same time base.

3.9.4 Interplanetary Cruise (Figure 3.9-4)

The cruise mode is established automatically by the guidance and control (G&C) subsystem subsequent to an acquisition. This mode utilizes the optical sensors and lag derived rate to control the nitrogen thrusters. Reacquisition of the celestial references, after inadvertent loss or occultation, is automatic.

During this phase, transmitter output is radiated over the low-gain antenna during early mission phases to approximately 100 days. The steerable high-gain antenna is employed thereafter. Prior to first mid-course maneuver, the high-power transmitter chain is activated and the launch transmitter turned off. Throughout the mission the receive function is maintained over the low-gain antenna configuration.

Cruise science data from the radiation detectors, magnetometer, and cosmic-dust detector is sequenced by the data automation equipment (DAE)

0.4
INTERPLANETARY CRUISE

NO.	EVENT	SOURCE	DESTINATION
1.	ACQUIRE ENGINEERING DATA	CAPSULE S/C SUB-SYSTEMS	TELEMETRY
2.	GATHER CRUISE DATA	DAE	SCIENCE
3.	PROCESS CRUISE SCIENCE DATA (DAE)	DAE	TELEMETRY
4.	TRANSMIT CRUISE SCIENCE CAPSULE & ENGINEERING DATA TO EARTH	TELE-METRY	RADIO
5.	RECORD SOLAR FLARE DATA	DAE	DATA STORAGE
6.	PROVIDE ONBOARD COMMAND TO PLAYBACK SOLAR FLARE DATA	DAE	DATA STORAGE
7.	PLAYBACK RECORDED SOLAR FLARE DATA	DATA STORAGE	RADIO
8.	MAINTAIN CELESTIAL REFERENCES (CRUISE)	C&S	G&C
9.	PROVIDE ON-SUN SPACECRAFT ENVIRONMENT CONTROL	TEMP. CONTROL	S/C
10.	SHIELD FLIGHT CAPSULE FROM SUN	S/C	-
11.	PROVIDE S/C SOLAR POWER	POWER	ALL S/S
12.	PROVIDE FLIGHT CAPSULE POWER	POWER	CAPSULE
13.	PROVIDE DOWNLINK COMMUNCATIONS	RADIO	DSN
14.	SWITCH TO HIGH POWER TRANSMITTER	C&S	RADIO
15.	PROVIDE UPLINK COMMUNICATIONS	RADIO	DSN
16.	SWITCH TO HIGH GAIN ANTENNA (80 TO 100 DAYS)	C&S	RADIO
17.	UPDATE HIGH GAIN ANTENNA POSITION	C&S	G&C
18.	UPDATE CANOPUS CONE ANGLE	C&S	G&C
19.	TURN OFF EARTH SENSOR (REMOVE EARTH SENSOR CANOPUS LOCK REQUIREMENT)	C&S	G&C
20.	RADIO PERIODIC SWITCHOVER	C&S	RADIO
21.	FLIGHT CAPSULE CANISTER SEPARATION (REFERENCE INFORMATION)	—	—

1 FUNCTIONS SHOWN THUS —| — DISCONTINUED DURING MANEUVER

2 BEFORE FIRST MID-COURSE TRAJECTORY CORRECTION

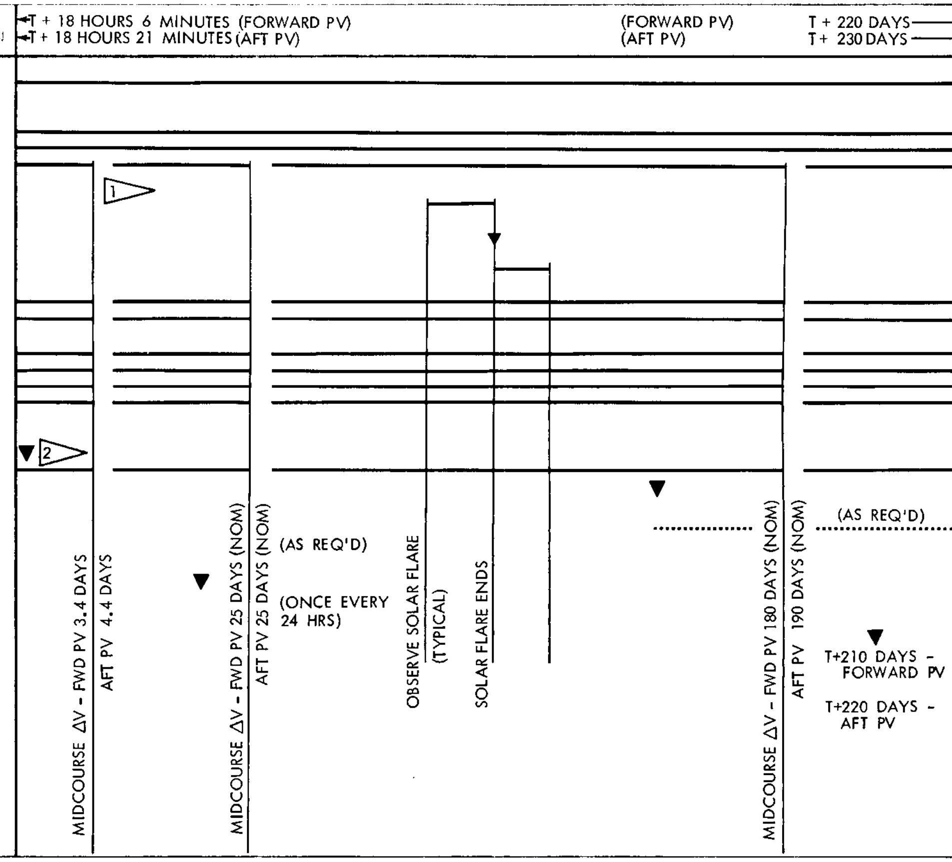


Figure 3.9-4: Flight Sequence — Interplanetary Cruise

throughout the cruise mode, and routed to the telemetry subsystem for real-time transmission to Earth. The flare-mode recorder is turned on by the DAE when a solar flare is detected.

3.9.5 Trajectory-Correction Maneuvers (Figure 3.9-5)

3.9.5.1 Application

A trajectory-correction maneuver consists of inertially (gyro) controlled turns which orient the thrust axis in the selected direction. This operation is followed by a midcourse engine propulsion-burn to achieve a velocity increment of a selected magnitude. Following the burn, the subsequent repositioning and celestial reference acquisition is automatic. Trajectory-correction maneuvers may be required during the following mission phases:

1) Interplanetary Trajectory Correction--Using onboard stored commands, the forward Planetary Vehicle will make an aiming point bias trajectory correction maneuver. The aft Planetary Vehicle will undertake the same maneuver and in addition will provide the maneuver for the required 10-day arrival separation time between Planetary Vehicles. The stored maneuver requirements can be updated by ground command.

2) Planetary-Vehicle Mars-Orbit Trim--Upon receipt of ground commands, the Planetary Vehicle performs the necessary trajectory correction maneuvers to adjust the orbital parameters to the required values. Following celestial reference reacquisition, the scan platform will be positioned and the planetary science acquisition program reinitiated.

0.5 INTERPLANETARY TRAJECTORY CORRECTION

NO.	EVENT	SOURCE	DESTINATION
1.	Provide Ground Maneuver Commands for Midcourse Correction (if required)	DSN	Radio Command C&S
	Receive and Store:	Radio/ Command	
	. Pitch (or Yaw) Turn Magnitude and Direction		
	. Roll Turn Magnitude and Direction		
	. Midcourse Motor Burn Time		
	. Velocity Increment		
	. Maneuver Start Time		
2.	Provide Stored Maneuver Commands	C&S	
3.	Maintain Celestial References		
	a. Canopus Roll Reference	Canopus Sensor	Reaction Control
	b. Sun Reference	Sun Sensor	Reaction Control
4.	Switch Maneuver Recorder ON	C&S	Data Storage
5.	Switch to Low-Gain Antenna	C&S	Radio
6.	Provide Inertial Reference-Roll Axis	C&S	G&C
7.	Roll to Required Attitude and Stabilize		
	Command Roll	C&S	Autopilot
	Roll Turn	Autopilot	Reaction Control
	Stop Roll Turn	C&S	Autopilot
	Stabilize	Autopilot	—
8.	Provide Inertial Reference Pitch and Yaw Axis	C&S	Autopilot
9.	Yaw (or Pitch) to Required Orientation		
	Command Turn	C&S	Autopilot
	Yaw or Pitch	Autopilot	Reaction Control
	Stop Turn	C&S	Autopilot
	Stabilize	Autopilot	—
10.	Provide ΔV		
	. Turn on Accelerometer		
	. Activate M.C. Engine	C&S	Pyro & Propulsion
	. Provide 2-axis TVC (Pitch and Yaw)	C&S	Autopilot
	. Start M.C. Engine	C&S	Pyro & Propulsion
	. Integrate Accelerometer Output	G&C	C&S

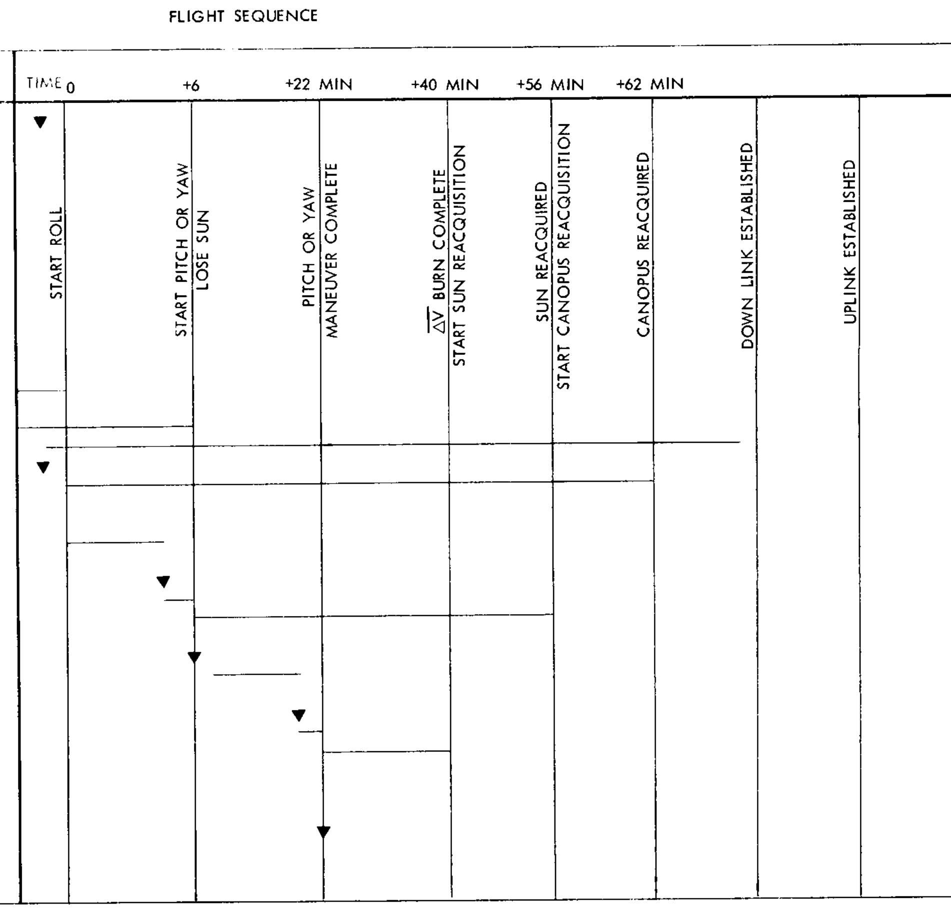


Figure 3.9-5: Flight Sequence — Interplanetary Trajectory Correction

0.5 INTERPLANETARY TRAJECTORY CORRECTION (Con't.)

NO.	EVENT	SOURCE	DESTINATION
10.	. Stop M.C. Engine	C&S	Propulsion & Pyrotechnic
	. Deactivate M.C. Engine	C&S	Propulsion & Pyrotechnic
	. TVC OFF	C&S	Autopilot
	. Stabilize P/V	G&C	
11.	Yaw (or Pitch) Back to Acquire Sun (Sequence of Commands)	C&S	Autopilot
12.	Command Celestial Reference	C&S	Autopilot
	a. Provide Sun Presence Signal	Sun Sensor	
	b. Switch to Sun Reference and Stabilize	G&C	Reaction Control
	c. Maintain Sun Reference	G&C	Reaction Control
13.	Roll Back to Canopus Line (Sequence of Commands)	C&S	G&C
14.	Provide Celestial Reference Command	C&S	G&C
	a. Sense Canopus	Canopus Sensor	G&C
	b. Switch to Canopus Reference	G&C	
	c. Maintain Canopus Reference	Canopus Sensor	Reaction Control
	d. Maneuver Recorder OFF	C&S	Data Storage
	e. Accelerometer OFF	C&S	G&C
	f. Switch to High-Gain Antenna (80 days on)	C&S	Radio
15.	Provide Communication Transmission Time Log Phase Lock-In Time	Radio	DSN
16.	Provide Uplink Communications Transmission Time Log Phase Lock-In Time	DSN	Radio
17.	Acquire and Process Engineering and Cruise Science Data	Science & DAE	Data Storage Subsystem
18.	Playback Recorded Data	C&S	Data Storage
19.	Telemeter Engineering and Science Data to Earth	Telemetry	Radio
20.	Shield Flight Capsule from Sun	S/C	F/C
21.	Provide S/C Off-Sun Environmental Control	Temp. Control	S/C
22.	Provide Solar Power to S/C . Sense Sun Presence . Provide Solar Power . Charge Battery	Power	Automatic

FLIGHT SEQUENCE

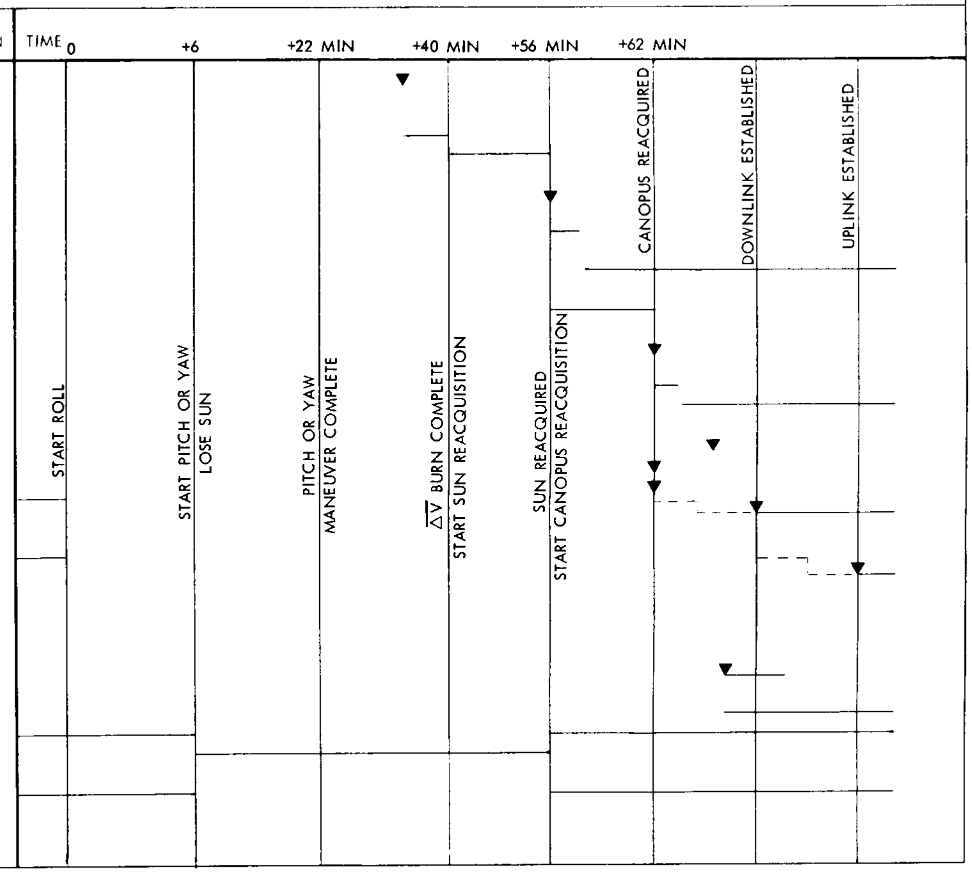


Figure 3. 9-5: (Continued) Flight Sequence — Interplanetary Trajectory Correction

0.5 INTERPLANETARY TRAJECTORY CORRECTION (Con't.) — FLIGHT SEQUENCE

NO.	EVENT	SOURCE	DESTINATION
23.	Provide Battery Power to S/C	Power	All Subsystem
24.	Provide On-Sun S/C Environmental Control	Temp. Control	S/C
25.	Shield Flight Capsule from Sun	S/C	F/C
26.	Provide Power to Flight Capsule	Power	Capsule

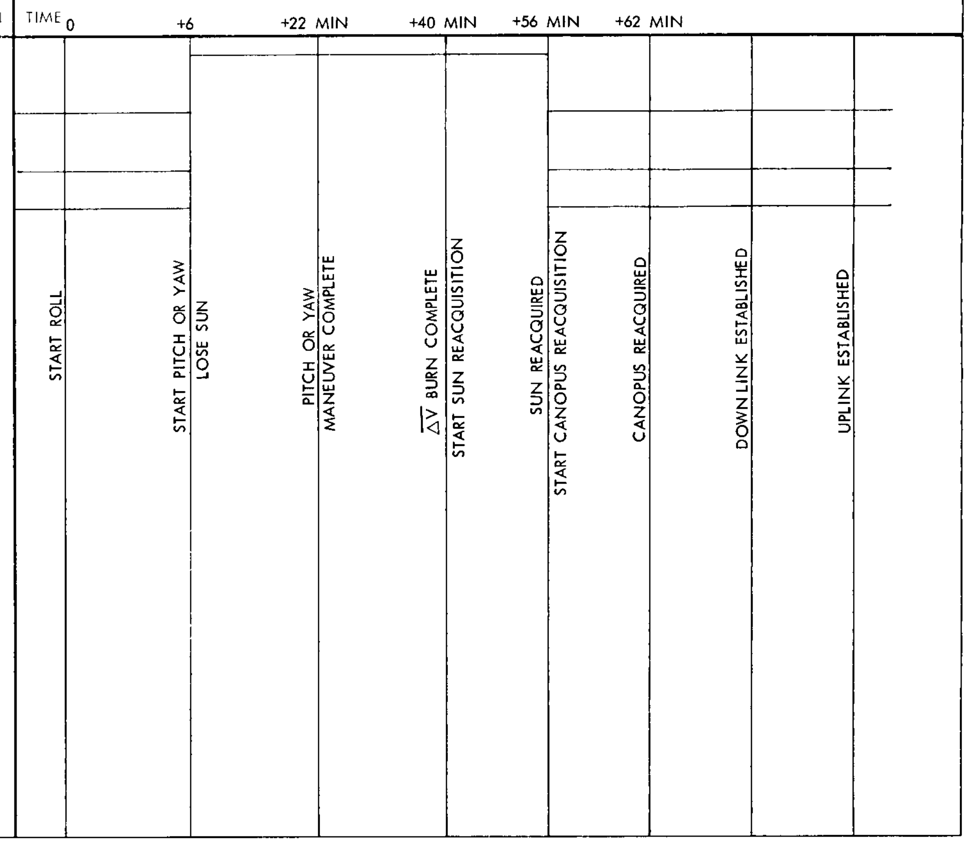


Figure 3.9-5: (Continued) Flight Sequence — Interplanetary Trajectory Correction

3) Flight Spacecraft Orbit Trim--Upon receipt of ground commands, the Flight Spacecraft performs the necessary trajectory correction maneuvers to readjust the orbital parameters if deflections result from capsule separation or other long-term perturbations.

3.9.5.2 Sequence

If a maneuver is required, the DSIF transmits a signal, which is received and demodulated in the radio subsystem and sent to the command subsystem for detection, decoding, and routing to the C&S subsystem. The required command data is: roll turn magnitude and direction, yaw or pitch turn magnitude and direction, ΔV magnitude, thrust termination time for backup (if the accelerometers malfunction), and maneuver start time.

The autopilot uses both position and rate from the inertial reference unit to control the N_2 thrusters. This mode is common to all axes. Maneuvering is accomplished by sequentially feeding a precision pre-timed rate command into the gyro-rate integrator of the roll and either the pitch or yaw axes. Proportional TVC control is obtained in all axes using autopilot commands and maneuver-engine jet vanes.

During off-Sun periods, spacecraft power will come from batteries. During trajectory-correction maneuvers in the late cruise phase, the radio subsystem is switched to the low-gain antenna to minimize loss of signal.

Spacecraft engineering data, capsule engineering data, and cruise science data will be accumulated during maneuvers by the data storage subsystem and reproduced later at a constant rate for inclusion in the master data format. Reproduced maneuver data will have priority over reproduced cruise science data. After the maneuver is completed, it is necessary to allow for a downlink phase lock-in time before the stored data is played back. Additional transmission time lag and phase lock-in time must be allowed before the uplink command capability is re-established.

3.9.6 Flight Capsule Canister Separation

The Flight Capsule biological barrier will be separated from each Planetary Vehicle prior to orbit insertion. Separation of the forward half of the canister from the Flight Capsule will be done by the Flight Capsule upon a stored command from the spacecraft. Canister separation time can be updated by ground command. Ground commands will be sent to reorient the spacecraft before canister separation, if required, in which case the down link may be broken and the time-lag and phase-lock sequences for the trajectory correction maneuvers of Section 3.9.5 would also apply.

3.9.7 Mars-Orbit Insertion

Each Planetary Vehicle orbit insertion will occur within view of Goldstone. The guidance scan platform assembly will track Mars during the planet approach phase to improve navigation accuracy prior to injection. An encoder in the scan-platform pointing mechanism will deliver pointing angle information to the telemetry subsystem for transmission to Earth.

The orbit-insertion-phase sequence is initiated by an on-board command. The requirements for maneuvers, propulsion burn, and reference reacquisition are functionally the same as for an interplanetary trajectory correction. Nominal maneuver times are as follows:

Maneuver	Time (Minutes)	
Roll (+140 deg)	12	
Stabilize	1	
Yaw (-35 deg)	3	Off-Sun 10.5 Minutes
Stabilize	1	
Rocket Burn	1.5	
Stabilize	1	
Yaw Back (+35 deg)	3	
Stabilize	1	
Roll Back (-140 deg)	12	
Stabilize	1	
Total	36.5	

The stored maneuver and velocity-increment commands can be updated by ground control. Pitch and yaw thrust-vector control is provided by Freon secondary injection into the exhaust stream of the solid-propellant orbit-insertion engine. Roll is controlled by activating a set of high-level thrusters using N_2 from the secondary injection pressurant storage. Angular position and rate information for all axes is obtained from the inertial reference unit. Following orbit insertion, the Planetary Vehicle performs the necessary maneuvers to reacquire the celestial attitude references. The high-gain antenna is pointed to illuminate Earth during orbit-insertion-rocket burn, thus providing data during insertion. Flight Spacecraft, cruise science, and capsule data are also recorded during the maneuver for later playback.

3.9.8 Planetary Vehicle Orbital Operations (Figure 3.9-6)

After Mars-orbit insertion, the Planetary Vehicle will maintain Canopus and Sun stabilization while orbiting around Mars. The Canopus sensor cone angles will be updated as required by the C&S subsystem or ground command. The limb and terminator detectors, one on each side of the spacecraft, will detect both morning and evening terminators and limb crossing for the first few orbits. Subsequently the DSIF/SFOF will determine these crossings.

Depending on orbit constants, occultation may occur with respect to Canopus, Sun, and Earth. If Canopus is occulted, the loss of the Canopus presence signal will switch the roll axes to the inertial hold mode. If the Sun is occulted, the loss of the Sun-presence signal will switch the pitch and yaw axes to the inertial hold mode; also the spacecraft power source will switch from the solar panels to the batteries. Earth-occultation information must be programmed in advance to the C&S subsystem. Under C&S control, the data storage system will record the spacecraft engineering and science data for delayed transmission to Earth.

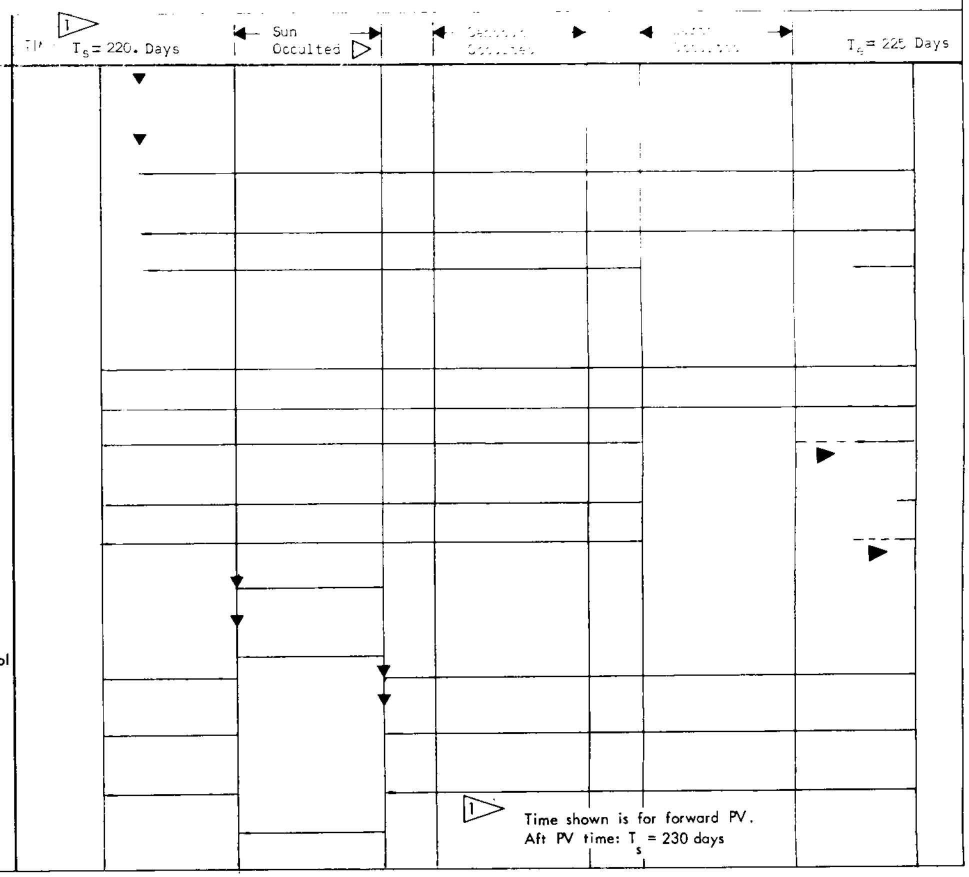
The radio subsystem functions as in the cruise mode during Mars orbit operations, except that the transmission of orbital information requires the high-gain antenna. The high-gain antenna (HGA) pointing control assembly responds to C&S programmed commands and points the high-gain antenna towards Earth. The C&S also commands, through the DAE, the off-on cycles of the science instruments.

Mode 2 is the primary data mode for this mission phase. The lower subcarrier is modulated with a composite PCM data train containing spacecraft-engineering, capsule-engineering, cruise-science, synchronization, and

0.8 PLANETARY VEHICLE ORBITAL OPERATIONS

FLIGHT SEQUENCE

NO.	[illegible]		[illegible]
1.	Provide Onboard Command-Initiate Science Orbital Operations Sequence	C&S	DAE
	a. Deploy Planetary Scan Platform	C&S	Pyro
2.	Acquire Orbital Science Data	DAE	Science Instruments
3.	Record Orbital Science Data	Science	Data Storage
4.	Playback Stored Science Data	DAE (C&S)	Data Storage
		Data Storage	Telemetry
5.	Acquire Capsule Engineering Data	F/C	Telemetry
6.	Acquire S/C Engineering Data	S/C	Telemetry
7.	Transmit Data to Earth (Mode 2)	Telemetry Radio	Radio DSN
8.	Provide Ground Updating Commands When Req'd	SFOF	DSN
9.	Provide Uplink Communications	DSN	Radio/Command
10.	Sense Sun Occultation	Sun Sensor	G&C
11.	Switch to Inertial Reference—Pitch & Yaw Axes	G&C (C&S Backup)	Automatic
12.	Inertial Hold—Pitch & Yaw Axes	Autopilot	Reaction Control
13.	Sense Sun Presence	Sun Sensor	G&C, C&S
14.	Reacquire Sun	G&C (C&S Backup)	Automatic
15.	Maintain Sun Reference	Autopilot	React. Control
16.	Provide Solar Power to Spacecraft	Power	
17.	Provide Battery Power to Spacecraft	Power	



▷ If required, depending on orbit selected.

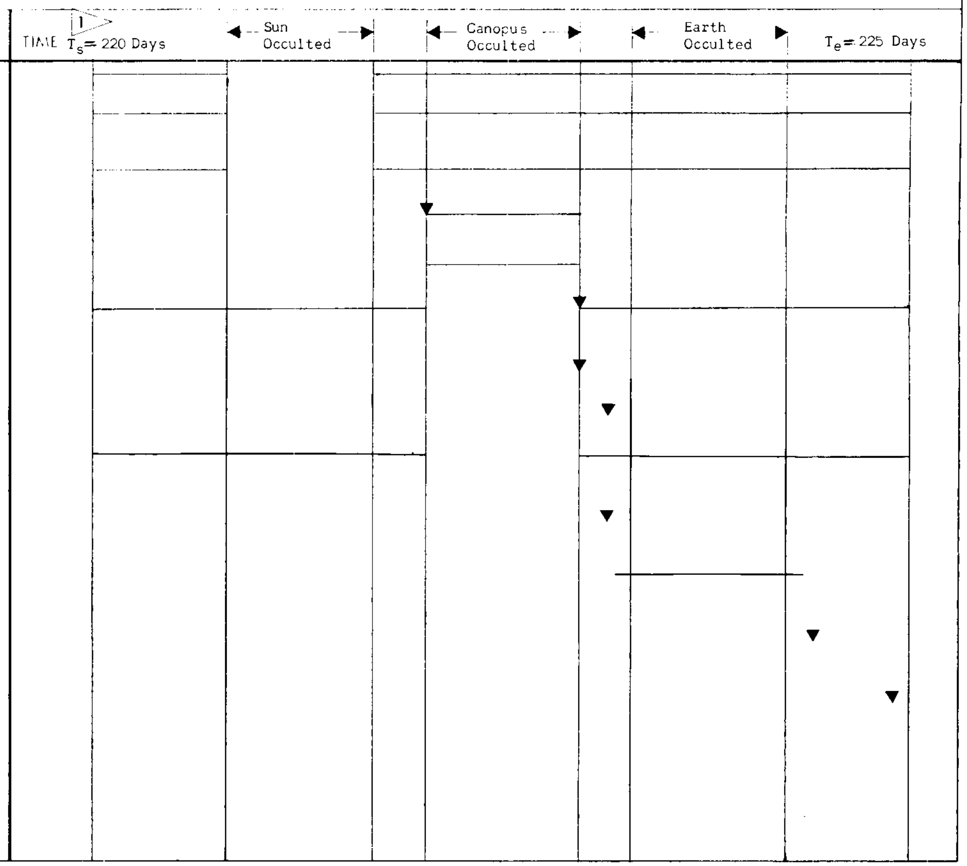
▶ Propogation & phase lock time lag

Figure 3.9-6: Flight Sequence — Planetary Vehicle Orbital Operations

0.8 PLANETARY VEHICLE ORBITAL OPERATIONS

FLIGHT SEQUENCE

NO.	EVENT	SOURCE	DESTINATION
18.	Provide Power to Flight Capsule	Power	F/C
19.	Provide On-Sun Spacecraft Environmental Control	Temp. Control	S/C
20.	Shield Capsule From Sun		
21.	Sense Canopus Occulted	Canopus Sensor	G&C, C&S
22.	Provide Inertial Reference Roll Axis	G&C (C&S Backup)	Automatic
23.	Provide Canopus Present Signal	Canopus Sensor	G&C, C&S
24.	Provide Celestial Reference Command	(G&C (C&S Backup)	Automatic
25.	Switch to Canopus Reference	G&C	Automatic
26.	Maintain Canopus Reference	Autopilot	Reaction Control
27.	Provide Onboard Initiate Earth Occultation Mode Sequence	C&S	Telemetry Data Storage
28.	Record Engineering & Science Data Onboard	Science & DAE	Data Storage
29.	Provide Onboard Termination of Earth Occultation Mode Sequence	C&S	Data Storage Telemetry
30.	Provide Ground Backup; Terminate Earth Occultation Sequence	SFOF DSN	Radio/Command



1 Time shown is for forward PV. Aft PV time: 230 to 236 days

Figure 3.9-6: Flight Sequence — Planetary Vehicle Orbital Operations (Continued) recorded data. The upper frequency subcarrier provides conveyance of planetary-science data to Earth from the temporary storage provided by the data storage subsystem. Included are data from each of the two TV cameras, IR scanner recorder, IR-UV spectrometer recorder, and capsule recorder.

3.9.9 Separation of Flight Spacecraft and Flight Capsule

Capsule separation will occur within view of Goldstone. Flight Capsule separation sequence will consist of gyro-controlled turns programmed by on-board command with ground-controlled updating, if required, to position the Planetary Vehicle to the required orientation. Signals for capsule control functions and for capsule separation are programmed by C&S stored commands and ground updating, if updating is required. Data Mode 2 can handle the increased activity during capsule checkout prior to release and subsequent relay of capsule engineering data after separation.

The capsule relay link is established prior to separation. After separation, capsule data is received by the radio subsystem. During the spacecraft maneuver for capsule separation, the spacecraft-DSN link will go out of phase-lock. The data received from the capsule is stored for transmission after ground reacquisition of the spacecraft signal. After capsule separation, the Flight Spacecraft automatically reorients to reacquire the Sun and Canopus.

3.9.10 Capsule Operations

Flight Capsule operations consist of orbit maneuver, orbital descent, entry, and terminal descent.

After separation, the Flight Capsule performs the necessary maneuvers to achieve deorbit along a selected Mars-impact trajectory. During descent, the Flight Capsule will transmit data to the Flight Spacecraft for subsequent relay to Earth. Flight Capsule operations continue until Mars impact.

3.9.11 Flight Spacecraft Orbital Operations

After Flight Capsule impact, the Mode 2 data channels allocated to the capsule are available for use by the data automation equipment as additional science-data capacity. The Flight Spacecraft operational sequence is essentially the same as given in Section 3.9.8, except that references to capsule operations are no longer applicable. After approximately 80 days in orbit, Data Mode 3 is initiated to maintain telecommunication-link margins.

3.10 PREFERRED SPACECRAFT LAYOUT AND CONFIGURATION

This section describes the 1971 Voyager spacecraft and includes: (1) a general-arrangement drawing of its exterior and inboard profile of subsystem equipment arrangement, and a functional block diagram of the spacecraft; (2) the structural alignment requirements, spacecraft reference axes and planes, coordinate system, and mechanical alignment; (3) the evaluation of various designs leading to the preferred design.

3.10.1 Summary

Several spacecraft configurations were studied in Task B. These studies included examination of the following candidate propulsion systems: (1) solid-propellant insertion motor and liquid monopropellant system, (2) LEM descent propulsion, and (3) Titan IIIC transtage--modified and unmodified.

The solid-propellant motor with monopropellant midcourse and orbit-trim engines was chosen for the preferred spacecraft design. The propulsion trades and configuration evaluation leading to this choice is reported in Section 3.11 of this document and in Volume C.

Configuration Model 945-8055 was selected as the preferred design because it excelled in the following features.

1) Subsystems are modular, which allows their complete checkout prior to installation in the spacecraft.
2) Electronic assemblies and propulsion subsystems are easily accessible for installation, maintenance, and testing.
3) Electronic assemblies are versatile in size, location, and construction, which permits grouping of electronic functions for simple interfaces, installation, and testing. Thermal balance and center of gravity (CG) location are easily obtained and can be revised to meet new requirements.
4) The 6.5-foot-diameter high-gain antenna provides margin over requirements for system gain consistent with transmission rates. Growth capability exists to accommodate an 8- by 12-foot paraboloid antenna with no effect on spacecraft length.
5) The solar array of eight fixed and four deployable panels (total area of 316 square feet) provides good growth potential by adding one segment to each folding panel (an increase of 138 square feet) or by modifying existing tab geometry.
6) The configuration is versatile and adaptable to 1973, 1975 and 1977 missions. The spacecraft is designed so that a range of trajectories

and Mars orbits can be achieved with minor changes to the 1971 design. It is also adaptable for missions to other planets.

3.10.2 General Description of Preferred Design

The general arrangement of the preferred spacecraft design is shown in Figure 3.10-1. Boeing Model 945-8055 complies with JPL "Mission Specification" capsule and launch vehicle interfaces and requires a height 50 inches less than the allotted spacecraft dynamic envelope height. Primary system elements and their interfaces are discussed in Section 3.5. Figure 3.5-1 includes references to spacecraft subsystem interface details described in Volumes A and B.

The preferred design utilizes a 120-inch-diameter semimonocoque cylinder as primary structure extending between the Planetary Vehicle Adapter and Flight Capsule interfaces. The adapter, a semimonocoque shell, supports the Planetary Vehicle at the base of the 120-inch-diameter shell and carries the loading to the nose fairing. The Planetary Vehicle is separated from the adapter at the base.

Two circular frames, 37 inches apart, define an electronic-assembly bay located approximately midway on the exterior of the primary structure. Electronic assemblies are mounted between these frames. Each assembly is an entity and can be adjusted around the periphery during design to meet thermal-balance, CG, radio-frequency interference, and magnetic-field requirements. If required, thermal-control louvers will be on the exterior surface of each assembly. The design allows maximum access for installation, maintenance, repair, inspection, and test. Space is available

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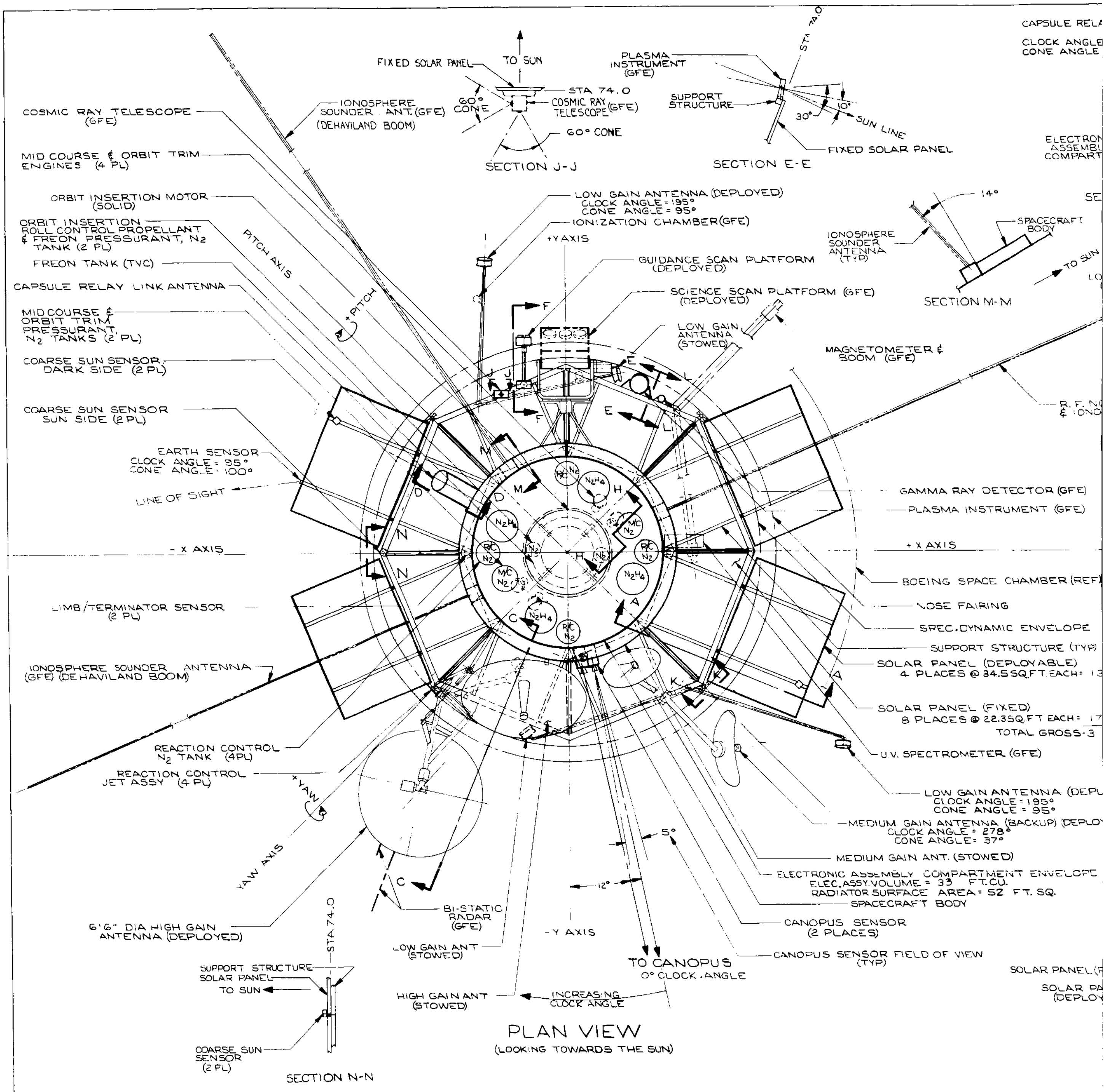
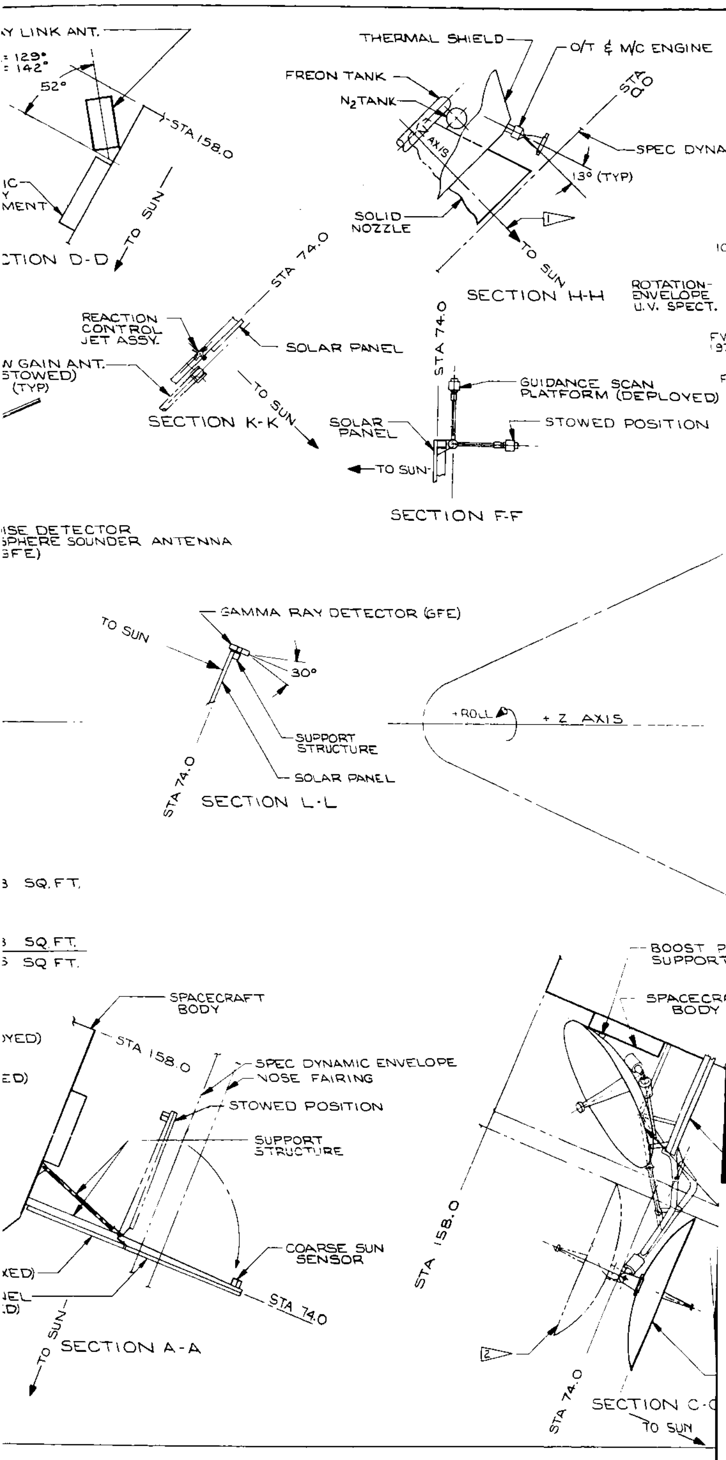


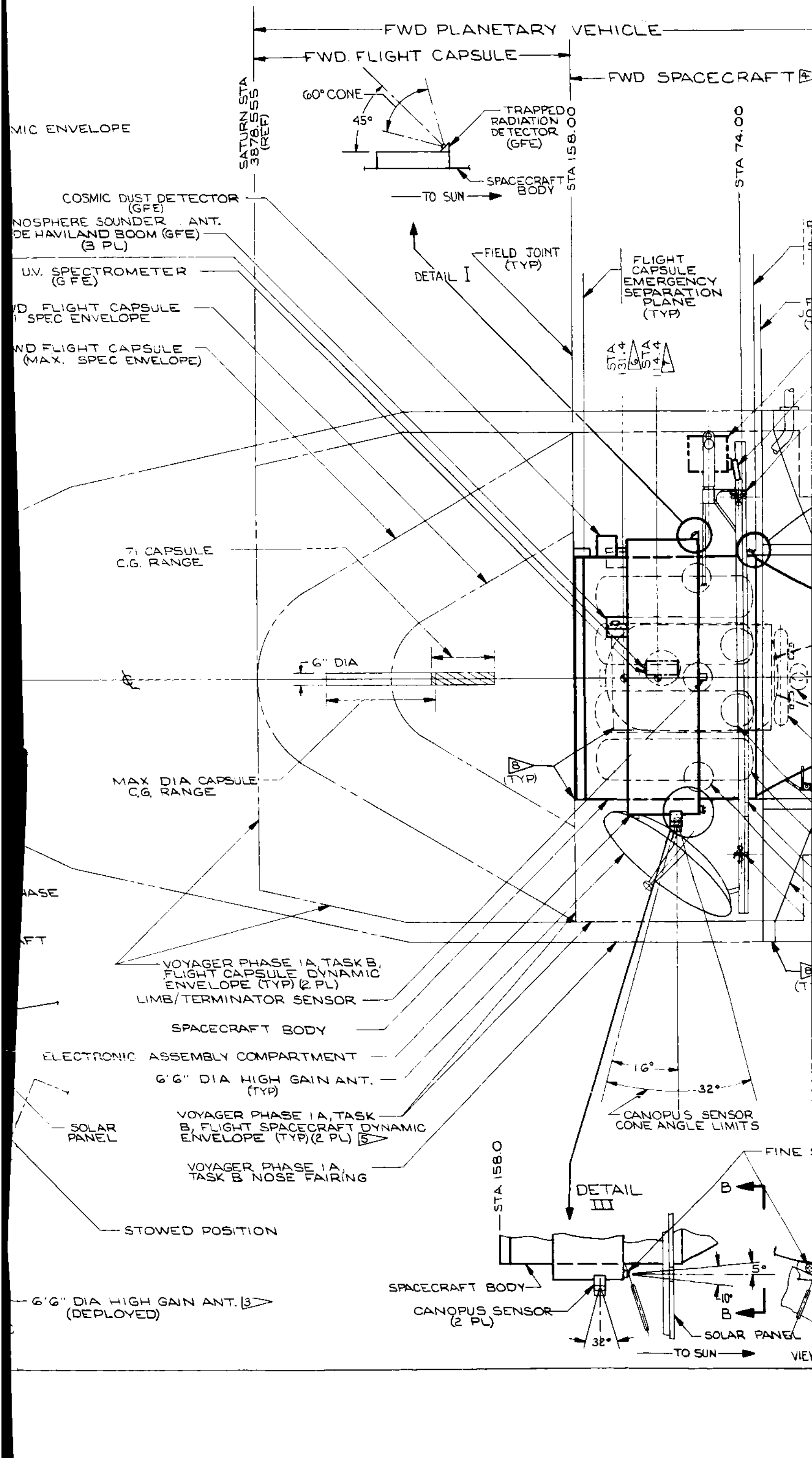
PLAN VIEW
(LOOKING TOWARDS THE SUN)

1



Y LINK ANT.
= 129°
= 142°
52°
STA 158.0
IC
Y
MENT
TO SUN
CTION D-D
THERMAL SHIELD
O/T & M/C ENGINE
FREON TANK
N2 TANK
AXIS
STA 0.0
SPEC DYNA
13° (TYP)
SOLID NOZZLE
1
TO SUN
SECTION H-H
ROTATION ENVELOPE U.V. SPECT.
STA 74.0
REACTION CONTROL JET ASSY.
SOLAR PANEL
N GAIN ANT. STOWED) (TYP)
TO SUN
SECTION K-K
STA 74.0
GUIDANCE SCAN PLATFORM (DEPLOYED)
SOLAR PANEL
STOWED POSITION
TO SUN
SECTION F-F
ISE DETECTOR
SPHERE SOUNDER ANTENNA
GFE)
GAMMA RAY DETECTOR (GFE)
TO SUN
30°
SUPPORT STRUCTURE
SOLAR PANEL
STA 74.0
SECTION L-L
+ROLL
+ Z AXIS
3 SQ. FT.
3 SQ. FT.
5 SQ. FT.
SPACECRAFT BODY
YED)
ED)
STA 158.0
SPEC DYNAMIC ENVELOPE
NOSE FAIRING
STOWED POSITION
SUPPORT STRUCTURE
COARSE SUN SENSOR
XED)
NEL
D)
STA 74.0
TO SUN
SECTION A-A
BOOST P
SUPPORT
SPACECRA
BODY
STA 158.0
2
STA 74.0
SECTION C-C
TO SUN

2



FWD PLANETARY VEHICLE
FWD. FLIGHT CAPSULE
FWD SPACECRAFT
SATURN STA 3878.555 (REF)
60° CONE
45°
TRAPPED RADIATION DETECTOR (GFE)
SPACECRAFT BODY
TO SUN
STA 158.00
STA 74.00
MIC ENVELOPE
COSMIC DUST DETECTOR (GFE)
NOSPHERE SOUNDER ANT. DE HAVILAND BOOM (GFE) (3 PL)
U.V. SPECTROMETER (GFE)
D FLIGHT CAPSULE SPEC ENVELOPE
WD FLIGHT CAPSULE (MAX. SPEC ENVELOPE)
DETAIL I
FIELD JOINT (TYP)
FLIGHT CAPSULE EMERGENCY SEPARATION PLANE (TYP)
71 CAPSULE C.G. RANGE
6" DIA
MAX DIA CAPSULE C.G. RANGE
(TYP)
VOYAGER PHASE IA, TASK B, FLIGHT CAPSULE DYNAMIC ENVELOPE (TYP) (2 PL)
LIMB/TERMINATOR SENSOR
SPACECRAFT BODY
ELECTRONIC ASSEMBLY COMPARTMENT
6'6" DIA HIGH GAIN ANT. (TYP)
VOYAGER PHASE IA, TASK B, FLIGHT SPACECRAFT DYNAMIC ENVELOPE (TYP) (2 PL)
VOYAGER PHASE IA, TASK B NOSE FAIRING
SOLAR PANEL
STOWED POSITION
6'6" DIA HIGH GAIN ANT. (DEPLOYED)
16°
32°
CANOPUS SENSOR CONE ANGLE LIMITS
DETAIL III
STA 158.0
FINE S
SPACECRAFT BODY
CANOPUS SENSOR (2 PL)
5°
10°
B
SOLAR PANEL
TO SUN

3

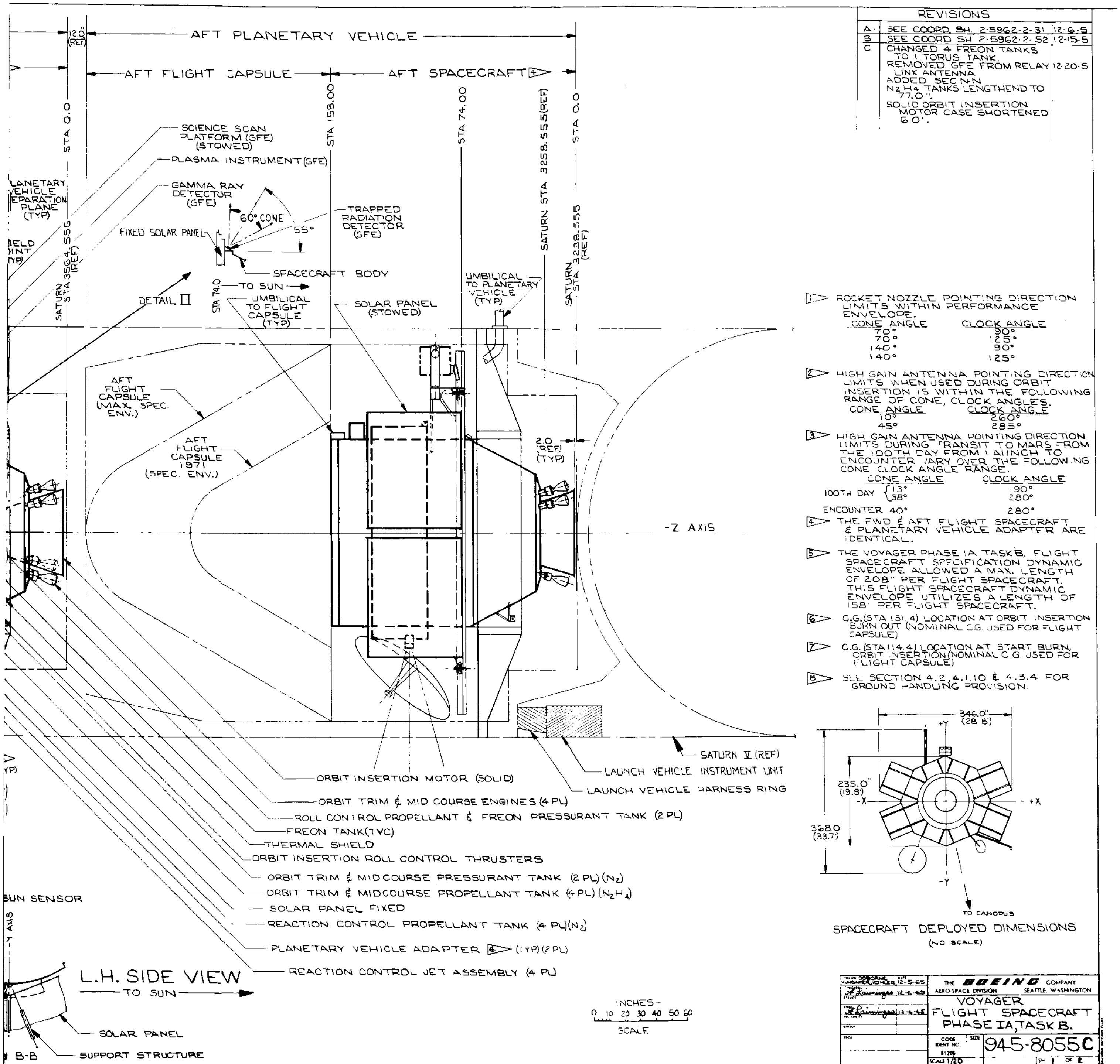


Figure 3. 10-1: Preferred Flight Spacecraft Model 945-8055

4

for 16 separate standard assemblies, each 16 inches wide, 32 inches high, and 7.8 inches deep. A usable volume of 33 cubic feet is provided in 14 assemblies. Two assemblies are available for growth. The 52-square-foot radiator area is approximately 50-percent greater than required to maintain a gross thermal balance. The packaging concept is described in Section 4.1.13.

The propulsion subsystem module, which can be assembled, bench-checked, leak-tested, or test fired as a complete unit prior to installation in the spacecraft, is located within the primary structure and attached to the base of the bus. This module consists of a solid-propellant orbit-insertion motor using Freon injection for thrust-vector control, four 200-pound-thrust monopropellant (N_2H_4) engines for midcourse and orbit-trim maneuvers, and two orbit-insertion roll thrusters. The motor nozzle is in the direction of the -Z axis. The midcourse and orbit-trim engines are located in pairs 45 degrees off the X and Y axes. The roll-control thrusters are located on the -Y axis. Four monopropellant tanks are located symmetrically about the principal control axes to maintain CG control. Tank pressurant (N_2) is stored in two spherical tanks.

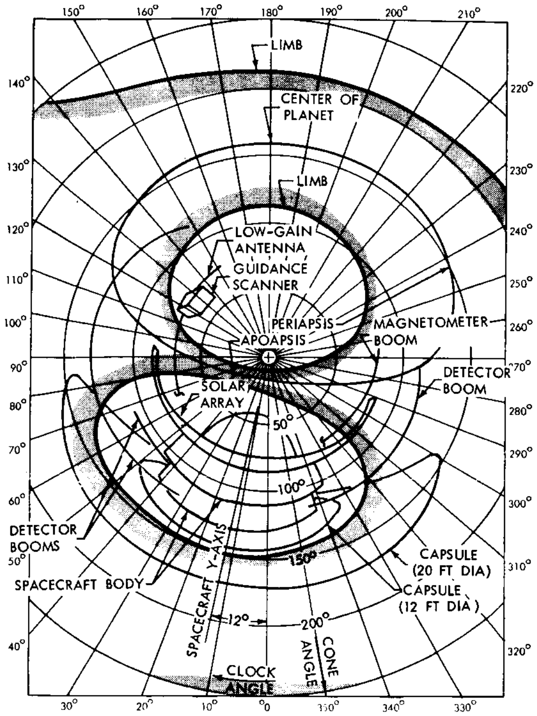
The reaction-control subsystem module uses cold gas (N_2) with coupled jet nozzles for attitude control. The four 0.25-pound-thrust nozzles are located on a 234-inch diameter, 45 degrees from the X and Y axes. Gas is supplied from four spherical tanks located on the X and Y axes. The reaction control subsystem is a two-part unitized system. Each half-section can provide the required torques, autonomously, if a component fails in the other half-section. This multichannel redundancy concept is used wherever possible throughout the design of the spacecraft subsystems and is discussed in detail in Section 4.1.2.

The solar panels are attached at the base of the primary structure to provide for equipment installation between the panels and Flight Capsule and to optimize back-face radiation of panel heat. The array consists of eight fixed trapezoidal panels (178 square feet) and four deployable rectangular panels (138 square feet). The deployable panels stow vertically on the dark side of the fixed panels. One segment added to each of the deployable panels will increase the area 138 square feet. Revising existing deployable panel shape will increase the area 46 square feet.

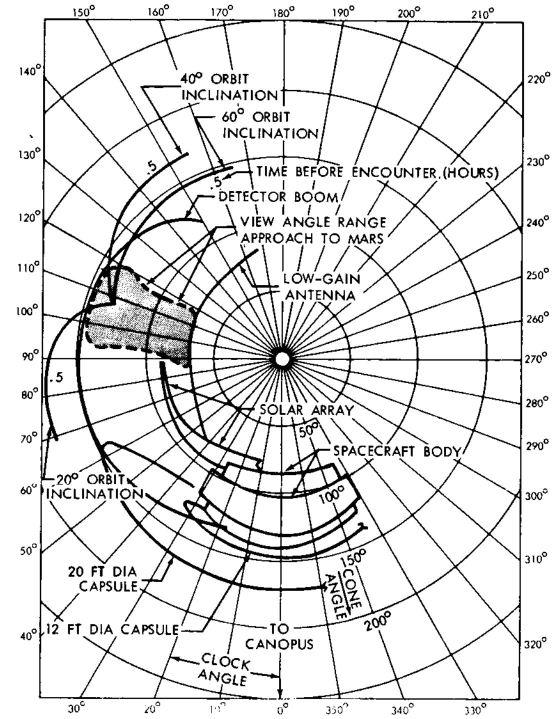
A 6.5-foot-diameter high-gain antenna with two rotation axes is located off the -Y axis and is stowed on the dark side of the solar panels. The primary and secondary axes are oriented to provide a minimum amount of rotation (Figure 3.10-2) and for maximum coverage of Earth during the orbit insertion burn, transit, and orbiting flight. An 8- by 12-foot parabaloid antenna (equivalent area of 10 feet in diameter) may be substituted without lengthening the spacecraft.

A Mariner C paraboloid antenna providing backup to the high-gain antenna is mounted on a boom in the +X, -Y quadrant. It is stowed on the dark side of the solar panels and is nonsteerable when deployed. The antenna is positioned to provide maximum coverage of Earth during the first month of Mars orbit.

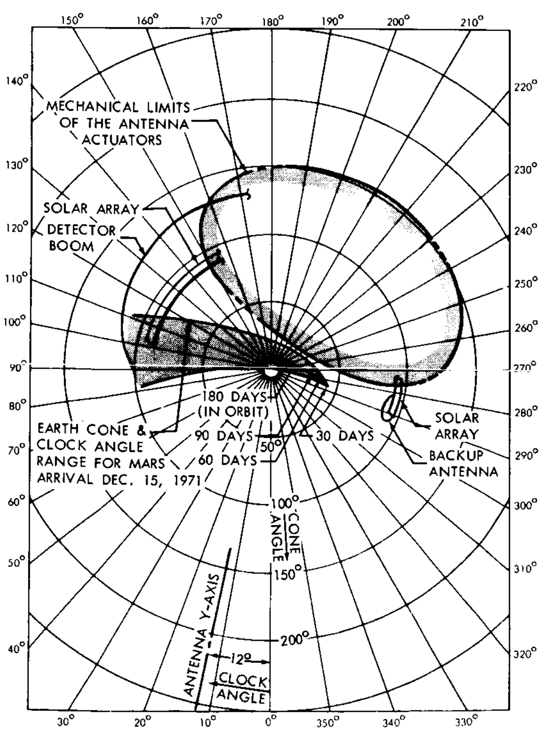
Two boom-mounted low-gain antennas are stowed on the Sun side of the solar panels. When deployed, these antennas are positioned in the -X +Y and +X -Y quadrants. One antenna functions as the launch and acquisition antenna, which must radiate through the nose fairing during the launch phase. A Flight Capsule relay antenna is located in the -X +Y quadrant and is bus-mounted on the dark side of the solar panels.



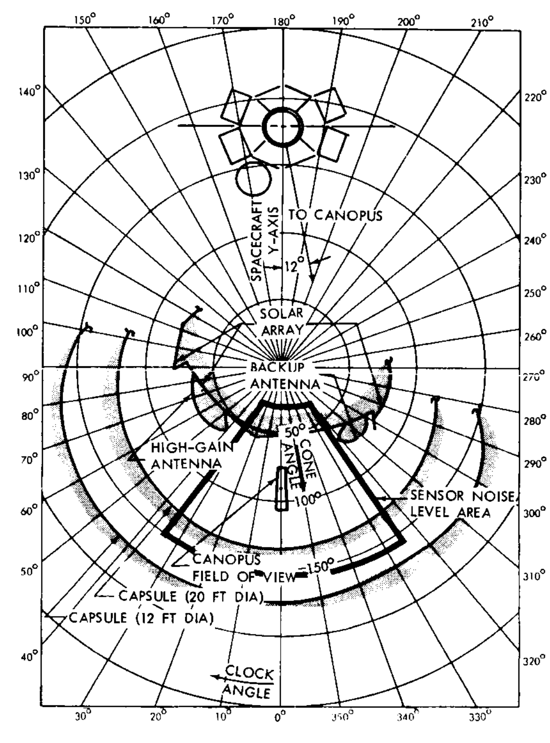
Science Scanner Shadowgraph (Toward Sun) lst Day Of Orbit



Guidance Scanner Shadowgraph (Toward Sun)



High-Gain Antenna Shadowgraph (Toward Sun)



Canopus Sensor Shadowgraph Voyager Trajectories (Toward Sun)

Figure 3.10-2: View Angles

A magnetometer (GFE) can be boom-mounted and located in the +X +Y quadrant when deployed. The boom is supported in a manner similar to the low-gain antennas.

The assumed science scan platform (GFE), stowed outboard of the electronic assemblies, is located on the +Y axis. The platform science equipment consists of an IR spectrometer, IR scanner, and two TV cameras. When deployed, the platform, which has a two-axis gimbal drive, can view the entire planet from any point in the orbit. The scan platform's field of view is shown in Figure 3.10-2. The UV spectrometer, with provision for two-axis motion, is mounted in an equipment bay. The assumed body- and boom-mounted science instruments (GFE) are described in Section 4.4 and located as shown in Figure 3.10-1.

The Canopus sensor is located in an electronic assembly 12 degrees counterclockwise from the -Y axis. This location provides the scan platform with maximum possible views of the planet while the spacecraft remains locked on its celestial references. The resulting spacecraft orientation provides the Canopus Sensor with proper views (Figure 3.10-2).

Redundant, fine Sun sensors are located on the base of the electronic assembly chassis containing the Canopus sensor and inertial-reference unit in the attitude-reference subsystem assembly. This grouping onto a single chassis reduces alignment tolerances to a minimum. Coarse Sun sensors are mounted on the Sun side of fixed solar panels located on the X axes. A second set of coarse Sun sensors is mounted on the dark side of the deployed solar panels and is used during initial acquisition of the Sun. Limb and terminator sensors are located on the X axis and are fixed on opposite sides and at the base of the electronic-assembly bay. A guidance scan platform is located off the +Y axis adjacent to the science scan platform. This platform with two rotation axes contains a planet tracker. Its field of view is shown in Figure 3.10-2.

The inboard profile of the electronic assemblies and other subsystems is shown in Figure 3.10-3. Installation of the electronic assemblies and subassemblies is defined in Section 4.1.13. Figure 3.10-4 is the functional block diagram.

3.10.2.1 Structural Alignments

Spacecraft design requirements include provisions for maintaining alignments during shipping, handling, boost accelerations and vibrations, and thermal deflections during flight.

The spacecraft coordinate system consists of three mutually perpendicular reference axes (Figure 3.10-5). The plane in which the X and Y axes lie is defined as the baseplane or Plane A. It is defined as spacecraft Station O (Saturn Station 3238.5) and spacecraft stations are numbered positively in a plus Z direction from this base. The Z axis is perpendicular to this plane and is located at the centerline of the spacecraft to coincide with the launch vehicle centerline. A secondary reference, Plane B, is established as being normal to reference Plane A and passing through the attitude reference subsystem mounting plate located on the -Y side of the spacecraft. This plane will be perpendicular to Reference Plane A within 0.20 milliradians.

The principal control axes are located 45 degrees from the X and Y spacecraft axes. Positive yaw is clockwise when the spacecraft is

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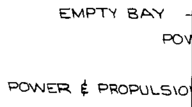
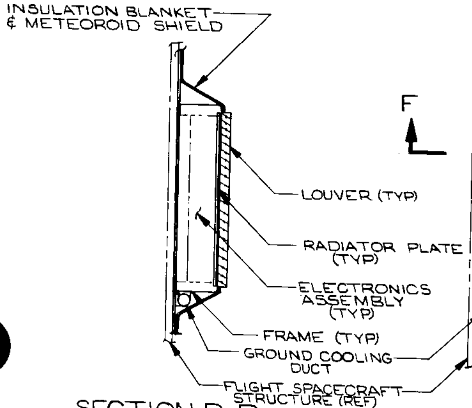
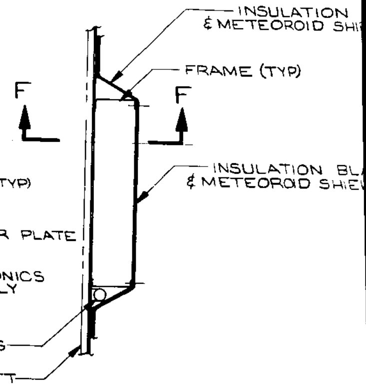




SECTION D-D
(1/10TH SCALE)

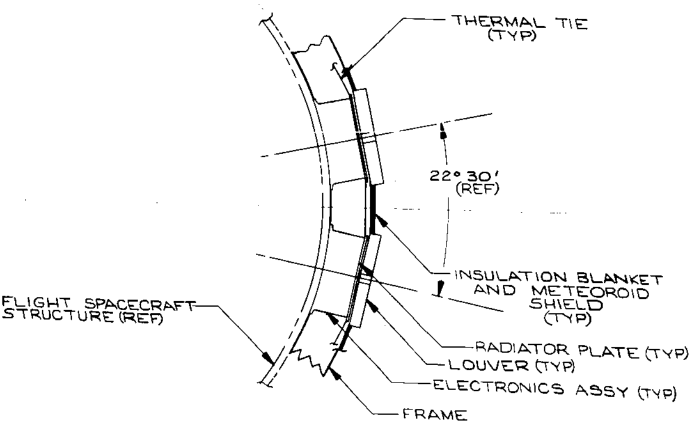
TO SUN



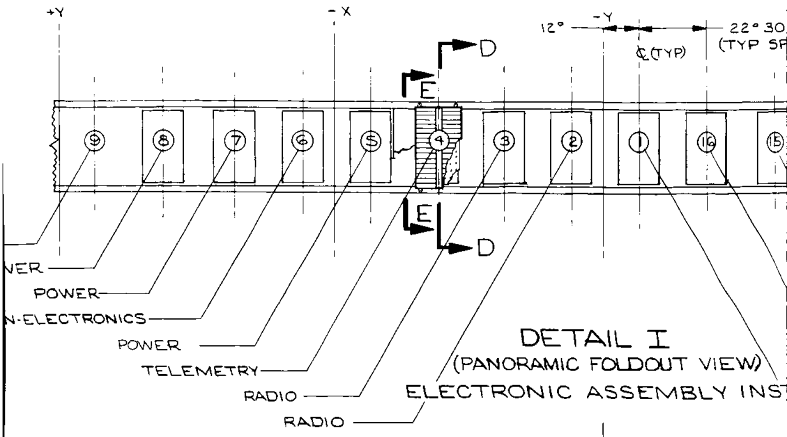
SECTION E-E
(1/10 TH SCALE)

TO SUN

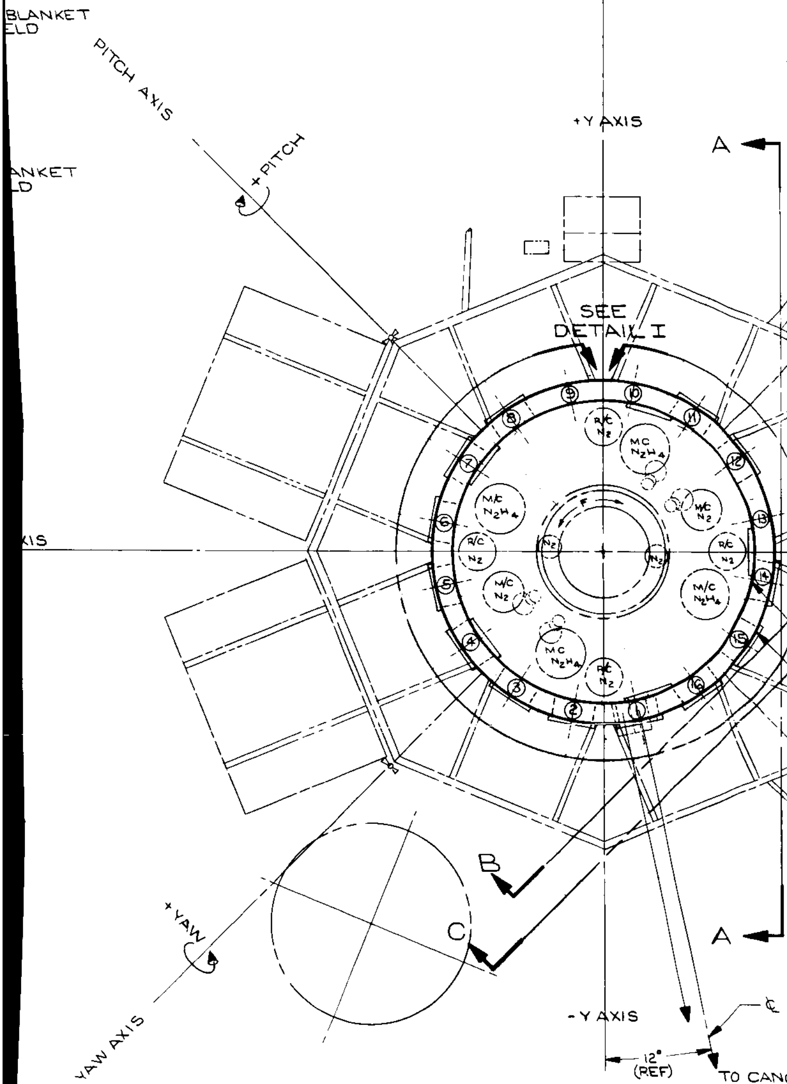
- X AX



SECTION F-F
(1/10TH SCALE)

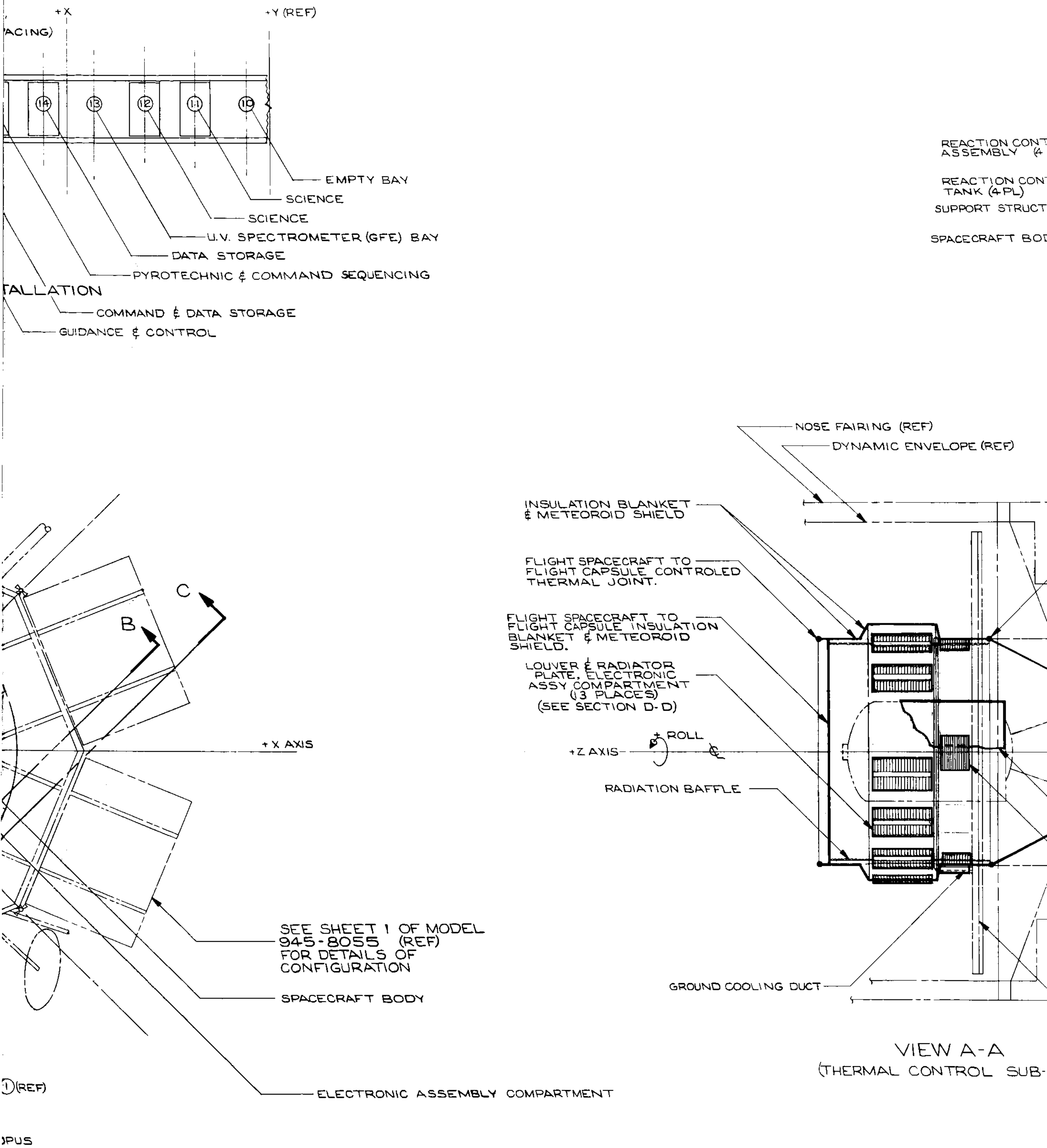


DETAIL I
(PANORAMIC FOLDOUT VIEW)
ELECTRONIC ASSEMBLY INST



PLAN VIEW
(LOOKING TOWARD THE SUN)

2


+X
+Y (REF)
ACING)
14
13
12
11
10
EMPTY BAY
SCIENCE
SCIENCE
U.V. SPECTROMETER (GFE) BAY
DATA STORAGE
PYROTECHNIC & COMMAND SEQUENCING
TALLATION
COMMAND & DATA STORAGE
GUIDANCE & CONTROL
REACTION CONT
ASSEMBLY (4
REACTION CONT
TANK (4 PL)
SUPPORT STRUCT
SPACECRAFT BOD
C
B
+X AXIS
SEE SHEET 1 OF MODEL
945-8055 (REF)
FOR DETAILS OF
CONFIGURATION
SPACECRAFT BODY
(REF)
ELECTRONIC ASSEMBLY COMPARTMENT
PUS
NOSE FAIRING (REF)
DYNAMIC ENVELOPE (REF)
INSULATION BLANKET
& METEOROID SHIELD
FLIGHT SPACECRAFT TO
FLIGHT CAPSULE CONTROLED
THERMAL JOINT.
FLIGHT SPACECRAFT TO
FLIGHT CAPSULE INSULATION
BLANKET & METEOROID
SHIELD.
LOUVER & RADIATOR
PLATE, ELECTRONIC
ASSY COMPARTMENT
(13 PLACES)
(SEE SECTION D-D)
+ ROLL
+Z AXIS
RADIATION BAFFLE
GROUND COOLING DUCT
VIEW A-A
(THERMAL CONTROL SUB-

3

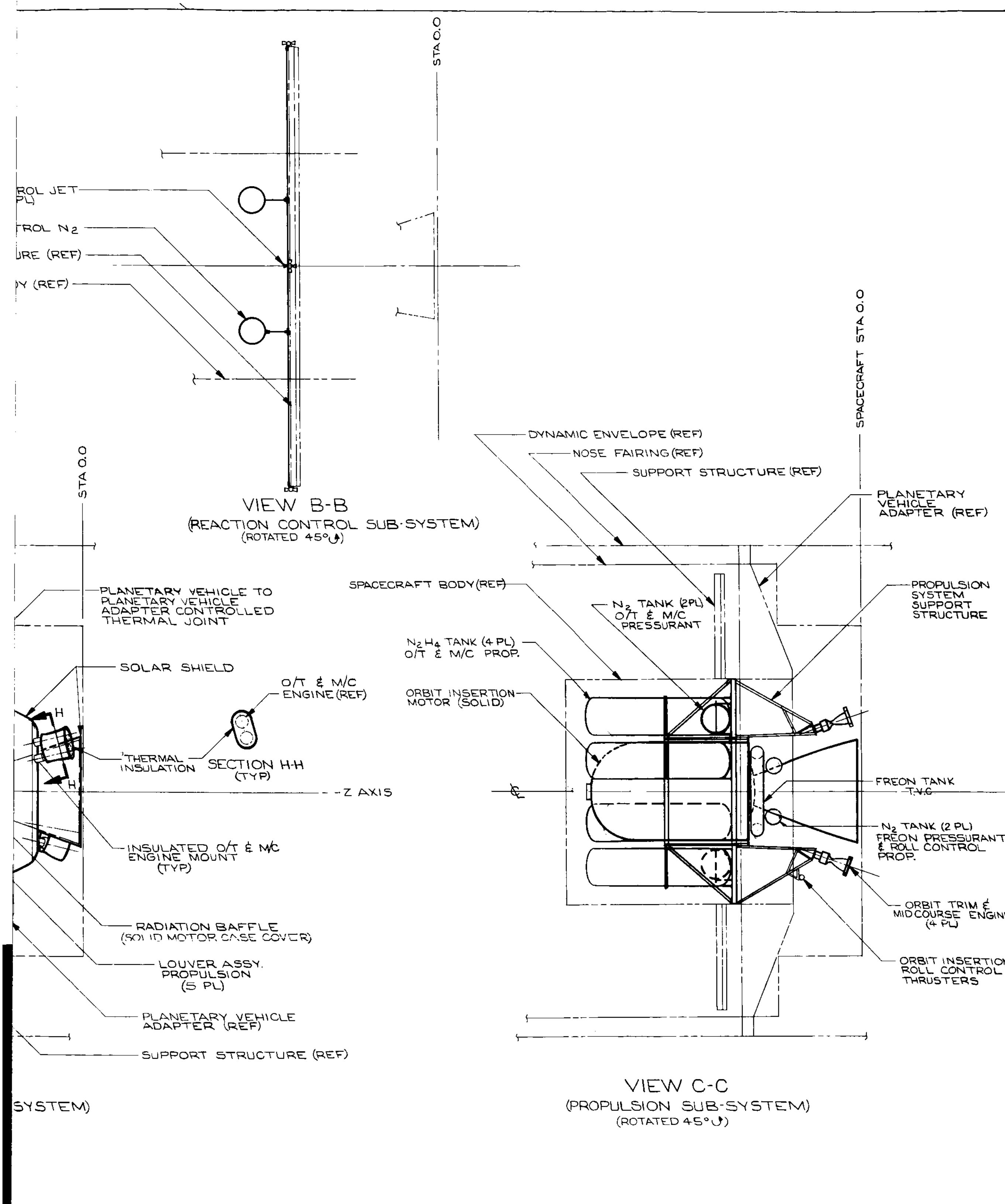
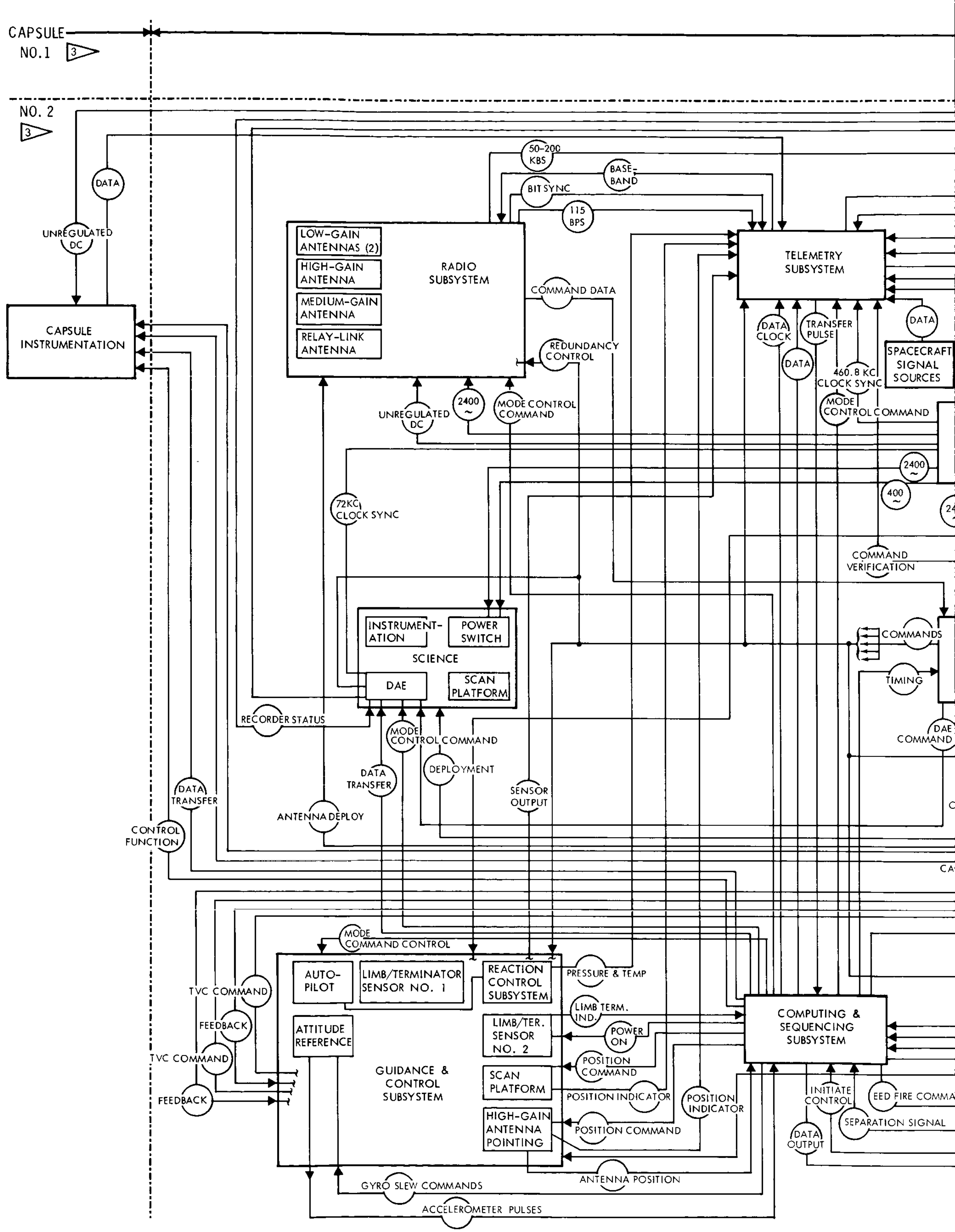
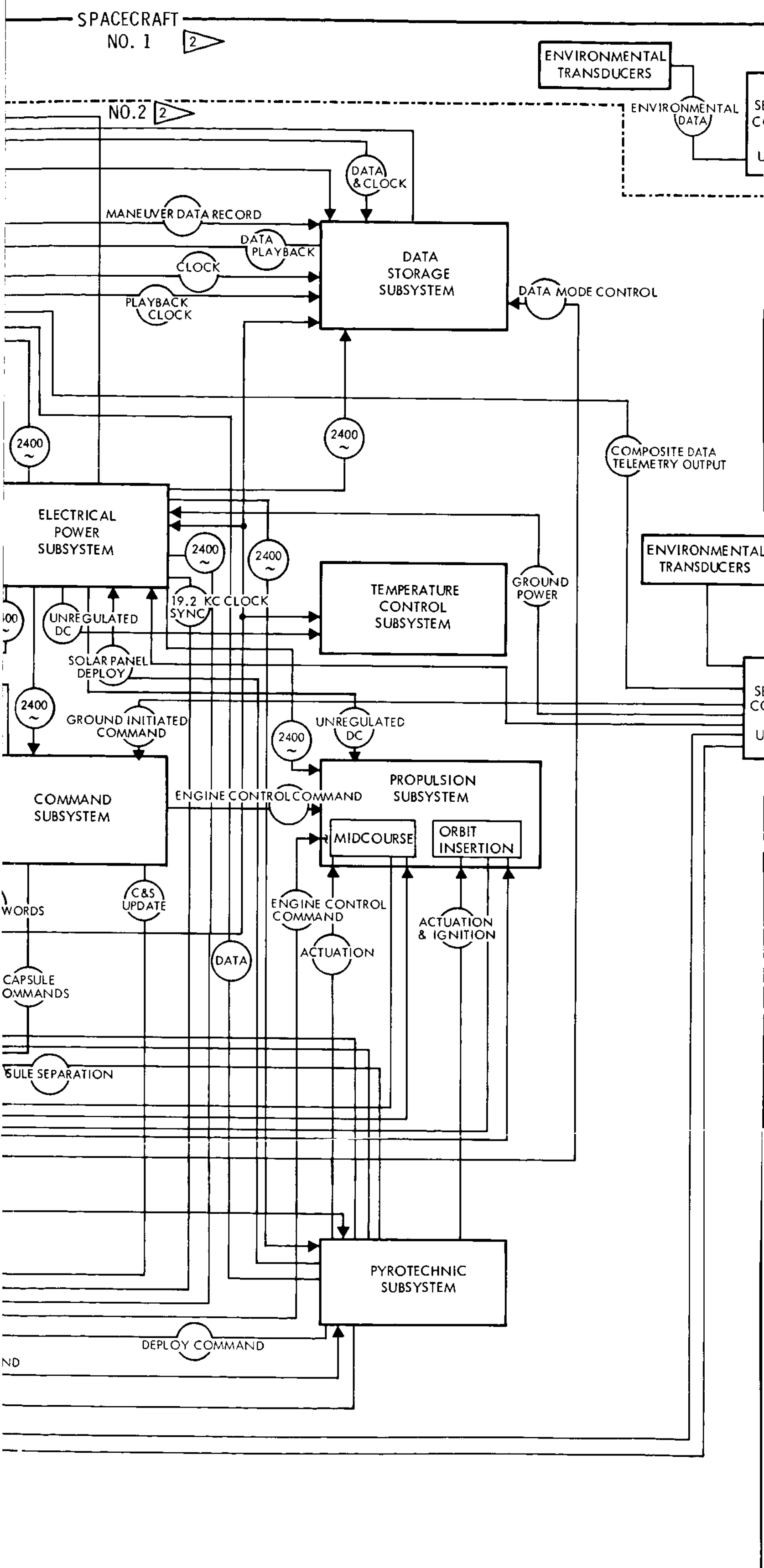


Figure 3.10-3: Inboard Profile — Flight Spacecraft

4



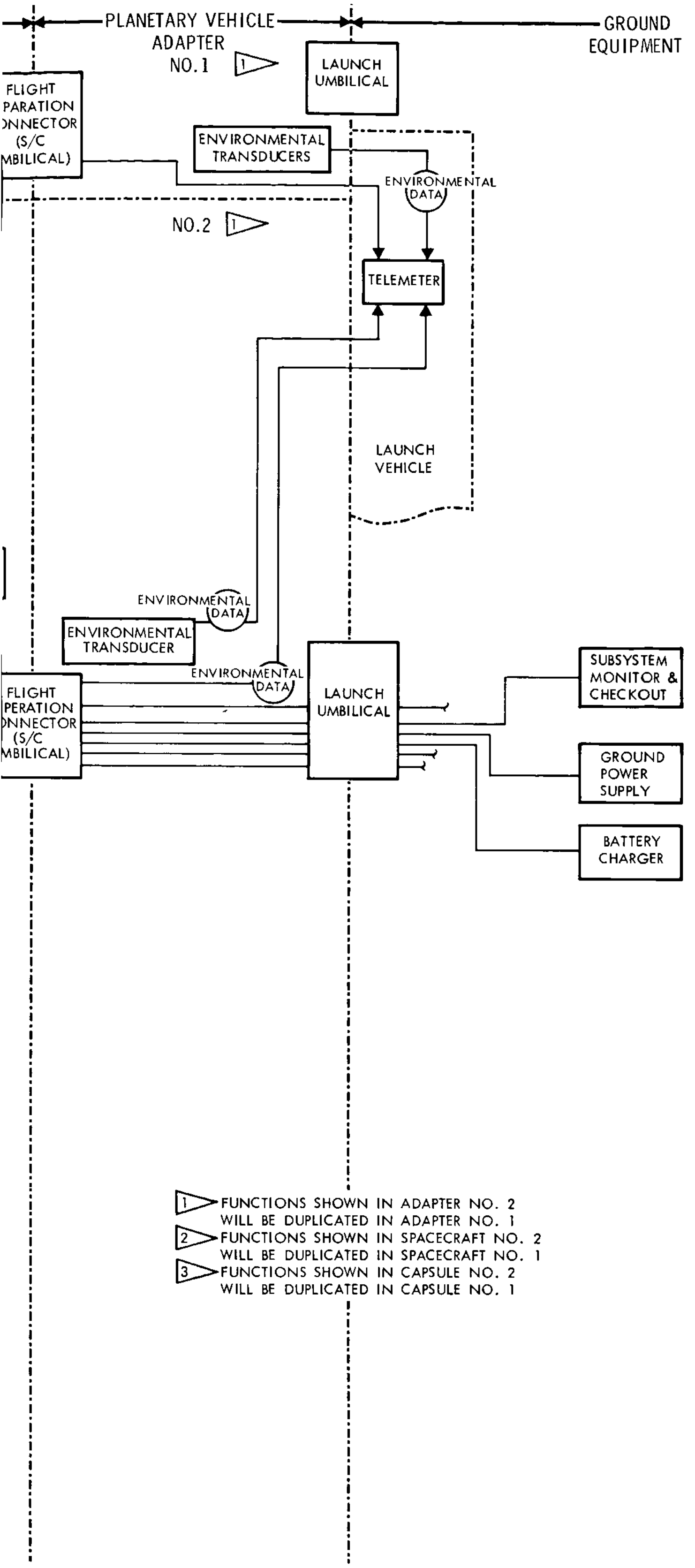
CAPSULE NO. 1
NO. 2
DATA
UNREGULATED DC
CAPSULE INSTRUMENTATION
50-200 KBS
BASE-BAND
BIT SYNC
115 BPS
LOW-GAIN ANTENNAS (2)
HIGH-GAIN ANTENNA
MEDIUM-GAIN ANTENNA
RELAY-LINK ANTENNA
RADIO SUBSYSTEM
COMMAND DATA
REDUNDANCY CONTROL
UNREGULATED DC
2400 ~
MODE CONTROL COMMAND
TELEMETRY SUBSYSTEM
DATA CLOCK
TRANSFER PULSE
DATA
460.8 KC CLOCK SYNC
SPACECRAFT SIGNAL SOURCES
MODE CONTROL COMMAND
2400 ~
400 ~
72KC CLOCK SYNC
COMMAND VERIFICATION
INSTRUMENT-ATION
POWER SWITCH
SCIENCE
DAE
SCAN PLATFORM
COMMANDS
TIMING
DAE COMMAND
RECORDER STATUS
MODE CONTROL COMMAND
DATA TRANSFER
DEPLOYMENT
SENSOR OUTPUT
ANTENNA DEPLOY
DATA TRANSFER
CONTROL FUNCTION
MODE COMMAND CONTROL
AUTO-PILOT
LIMB/TERMINATOR SENSOR NO. 1
REACTION CONTROL SUBSYSTEM
PRESSURE & TEMP
LIMB TERM. IND.
POWER ON
ATTITUDE REFERENCE
LIMB/TER. SENSOR NO. 2
POSITION COMMAND
TVC COMMAND
FEEDBACK
TVC COMMAND
FEEDBACK
GUIDANCE & CONTROL SUBSYSTEM
SCAN PLATFORM
POSITION INDICATOR
POSITION INDICATOR
HIGH-GAIN ANTENNA POINTING
POSITION COMMAND
COMPUTING & SEQUENCING SUBSYSTEM
INITIATE CONTROL
EED FIRE COMMA
SEPARATION SIGNAL
DATA OUTPUT
ANTENNA POSITION
GYRO SLEW COMMANDS
ACCELEROMETER PULSES



SPACECRAFT
NO. 1
2
NO. 2
2
ENVIRONMENTAL TRANSDUCERS
ENVIRONMENTAL DATA
DATA & CLOCK
MANEUVER DATA RECORD
DATA PLAYBACK
CLOCK
PLAYBACK CLOCK
DATA STORAGE SUBSYSTEM
DATA MODE CONTROL
2400 ~
2400 ~
COMPOSITE DATA TELEMETRY OUTPUT
ELECTRICAL POWER SUBSYSTEM
2400 ~
2400 ~
ENVIRONMENTAL TRANSDUCERS
GROUND POWER
TEMPERATURE CONTROL SUBSYSTEM
UNREGULATED DC
19.2 KC CLOCK
SYNC
SOLAR PANEL DEPLOY
2400 ~
GROUND INITIATED COMMAND
UNREGULATED DC
2400 ~
COMMAND SUBSYSTEM
ENGINE CONTROL COMMAND
PROPULSION SUBSYSTEM
MIDCOURSE
ORBIT INSERTION
WORDS
C&S UPDATE
ENGINE CONTROL COMMAND
ACTUATION & IGNITION
DATA
ACTUATION
CAPSULE COMMANDS
SULE SEPARATION
PYROTECHNIC SUBSYSTEM
DEPLOY COMMAND
ND

Figure 3.1(

2



)-4: Spacecraft Functional Block Diagram

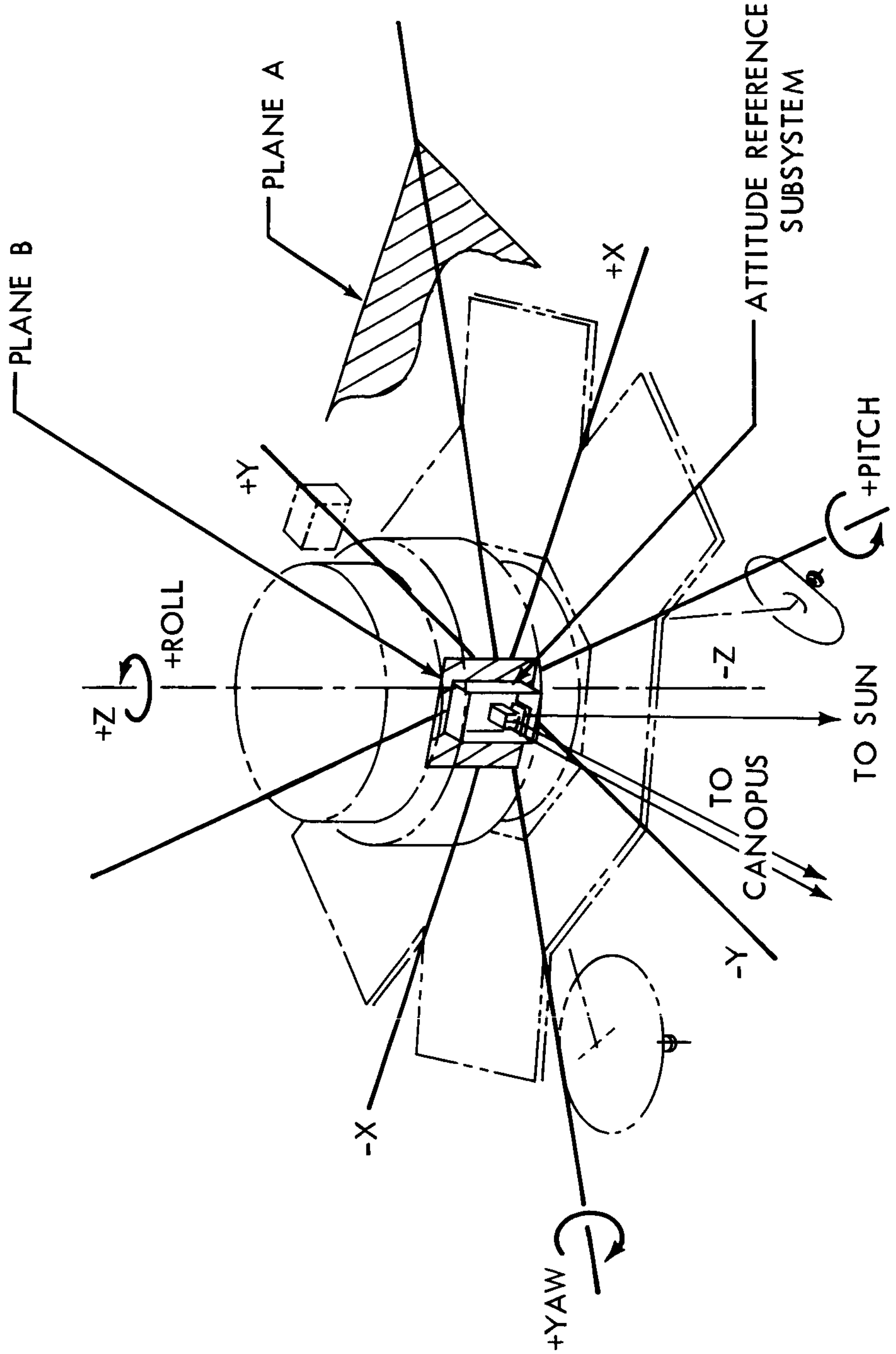


Figure 3. 10-5: Voyager Coordinate System

viewed from the -Y side of the spacecraft. Positive pitch is clockwise when the spacecraft is viewed from the -X side of the spacecraft. The z axis is the reference for roll, and positive direction is defined as clockwise when the spacecraft is viewed from the Sun side. An alignment summary chart for subsystem elements, sensors, and science instruments is shown in Table 3.10-1.

3.10.2.2 Interfaces

The Spacecraft Bus interfaces with the Planetary Vehicle Adapter subsystem, propulsion subsystem, and space science subsystem are discussed in Sections 4.2, 4.3, and 4.4, respectively. These subsystem elements and their interfaces are diagrammed in Figure 3.5-1.

3.10.3 Selection of the Preferred Design

This section describes the sequence of design and evaluation used to select the preferred design. The selection process is described below.

Review of Task A Configuration Against Task B Requirements--This review revealed major differences in mission constraints affecting the configurations. The Task A 1971 mission configuration features all deployable solar panels that were also usable in the 1969 test vehicle at a stowed envelope consistent with the Surveyor nose fairing on the Atlas-Centaur launch vehicle. This was done to meet the objective of maximum similarity between the 1971 Mars mission and the 1969 test mission configurations. This objective does not apply to Task B; therefore, a maximum amount of the 20-foot diameter was used for fixed panels.

Table 3.10-1: Alignment Summary Chart

Subsystem or Sensor	Location	Remarks
Attitude Reference	Fixed to spacecraft body -Y, +X quadrant	Align exactly with reference Plane B and parallel to reference Plane A within ±0.5 milliradian, guide pin vertical position from Plane A controlled within ±.025 inch
Canopus Sensor	Mounted in attitude reference subsystem package	Aligned to package mounting base within ±1.0 milliradian and oriented normal to reference Plane B and parallel to Plane A
Fine Sun Sensor	Mounted in attitude reference subsystem package	Aligned to package mounting base within ±2.0 milliradians
Inertial Reference Unit	Mounted in attitude reference subsystem package	Aligned to package mounting base within ±1.0 milliradian
Coarse Sun Sensors	Dark side of deployed solar panel	Aligned with reference Plane A ±0.5 degree
Coarse Sun Sensors	Sun side of fixed solar panels on the -X, +X axis	Aligned parallel to reference Plane A ±0.5 degree
Guidance Scan Platform	Mounted on +Y side with a 2-axis gimbal	Axis in X direction parallel to reference Plane A and normal to Plane B within ±2 milliradians
Limb/Terminator Sensor	Fixed to spacecraft body -X, +X axis	Aligned parallel to Plane A and X axis within 0.5 degree
Earth Sensor	Fixed to solar panel	Aligned to a cone and clock angle relative to 1 degree of spacecraft axes
Magnetometer Mounting	Extendable boom +X, +Y quadrant	Aligned with Z axis within 1 degree all directions
High-Gain Antenna	Deployable boom -X, -Y quadrant	Aligned to theoretically required angle within ±2 milliradians
Low-Gain Antenna	Boom mounted in -X, +Y and +X, -Y quadrants	Aligned to theoretically required cone and clock angle within ±1 degree
Solar Panels	Fixed to spacecraft body	Parallel to reference Plane A within ±0.5 degree
Reaction Control Nozzles	Located 45 degrees to the X and Y axis at a 234-inch diameter	Normal to reference Plane A within ±2 degrees
Propulsion -- Orbit Insertion	Along Z axis on spacecraft	Thrust axis passes through CG of Planetary Vehicle within 0.06 inch & parallel to Z axis within 2 milliradians
Propulsion -- Midcourse and Orbit Trim	Canted 13 degrees to orbit insertion engine on spacecraft	Thrust axis of each midcourse engine aligned with ±0.25 degree
Science Scan Platform Mounting	Mounted on +Y side with a 2-axis gimbal	Axis in X direction parallel to reference Plane A and oriented 12 degrees from reference Plane B within ±0.5 milliradian
Science Instrument Mountings	Boom mounted and fixed to spacecraft body	Interface alignments recognize sensor requirements

A Task A capsule was not carried into Mars orbit, which allowed the insertion motor to be oriented away from the Sun and the solar cells. In Task B, a much larger insertion motor is oriented so that the spacecraft/capsule combination is placed in Mars orbit. Considerations of CG control tend to favor orientations of the insertion thrust line along the capsule-spacecraft centerline; thus, the motor exit plane is toward the Sun.

Other Task B considerations that influenced spacecraft dimensions and arrangements are: (1) a 120-inch-diameter capsule interface (was 80 inches on Task A), (2) an approximate 21,000-pound planetary vehicle weight (was approximately 8,000 pounds on Task A), and (3) solar-panel area of approximately 320 square feet (was approximately 240 square feet on Task A).

Establish a Baseline Configuration Concept--This step provided a configuration used as a basis for variation of the major configuration features with subsequent analysis of the resulting designs. Properties such as weight and mass balance, thermal balance, sensor views, and spacecraft size were evaluated.

Configuration Evolution--Configuration evolution for the proposed Voyager spacecraft is shown in Table 3.10-2. A total of 29 spacecraft configurations were evaluated in gross terms. This evaluation emphasized enhancing features such as reduced spacecraft length, improved views, and reduced panel temperatures. Using these design features, nine additional configurations were developed and, from the original 29, two were selected for further evaluation. These 11 included the configurations used in the propulsion trades reported in Volume C.

Table 3.10-2: Configuration Evolution

PROPULSION TYPE	Boeing Config. Model Number	PV Adapter	Gross Conceptual Evaluation: Propulsion	Solar Panel	Views: Science	Canopus	Thermal	Telecom	Equipment Comp	Primary Structure	Weight	Center of Gravity	Second Level Evaluation	Detailed Evaluation	Preferred Design
LEM DESCENT	945-8010		X			●									
	945-8011	X		●		●			X						
	945-8012			X						X			Propulsion		
	945-8013		●							X			Trade		
TITAN TRANSTAGE	945-8020		X				X						Propulsion		
	945-8021		X	X			●						Trades		
	945-8022		X						X				945-8023		
	945-8024			X		●	X	●		X	●		945-8025		
	945-8026		X	X				X				●			
	945-8027		X								●	●			
	945-8028		●	X											
	945-8029		X				X						945-8029A		
SOLID MOTOR	945-8030		●	●	X				X				945-8033		
	945-8031		X				●		X				Propulsion	945-8033	
	945-8032		X	X					X				Trade		
	945-8034			X		●	●			X			945-8035		
	945-8036				X		●		X				945-8051	945-8055	945-8055
	945-8037				●				X						
	945-8038			X	●	●		●							
	945-8039			X		●									
	945-8040			X		X							945-8043	945-8045	
	945-8041			X	X									945-8054	
	945-8042			X		●									
	945-8044						X		X		●				
	945-8046	X		●						X			945-8049		
	945-8047	X							X				945-8053		
	945-8048	X							X	●					
	945-8050	X		●				●		X					
	945-8052			X			●		X						

NOTE: X = Varied Feature
● = Elimination Feature

Four candidates using the selected propulsion concept were chosen for detailed evaluation.

Selection of Preferred Design--Four candidate configurations were evaluated using methods described in Section 3.10.3.2. Boeing Model 945-8055 was selected and is the preferred spacecraft system described in this report.

3.10.3.1 Candidate Spacecraft Configurations

In addition to the preferred configuration described in 3.10.2, three other spacecraft configurations were selected for detailed evaluation. The first of these, Model 945-8045 (Figure 3.10-6), is similar to the preferred design except for solar-panel location, shape, and ratio of fixed solar-panel area to deployed solar-panel area. Another difference is the location of the electronic assemblies on the Sun side of the solar panels as opposed to those of the preferred design, which are located on the dark side of the solar panels. The high-gain antenna is rotated 180 degrees from its location on the preferred configuration. The spacecraft is 9 inches shorter and weighs 22 pounds less than the preferred design.

The second candidate configuration, Model 945-8054, is shown in Figure 3.10-7. This configuration differs from the preferred design in that it has a conical primary structure rather than a cylindrical body, a reduced diameter of the electronic assembly bay (80 inches versus 120 inches) and a reduced potential radiator area, a Sun-side location of electronic assemblies, different solar-panel location and shape, and different ratio of fixed area to deployed area. The Planetary Vehicle

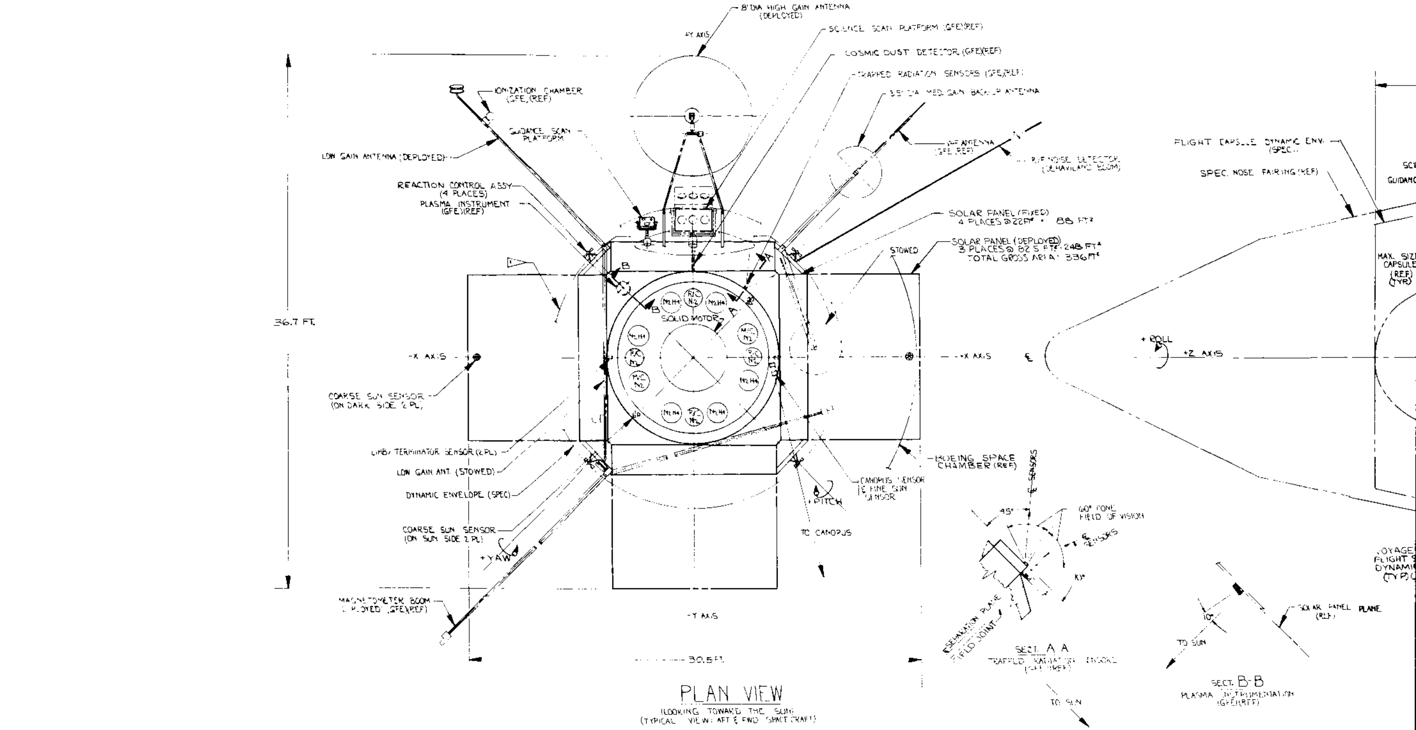


Figure 3.10-6: Model 945-8045,

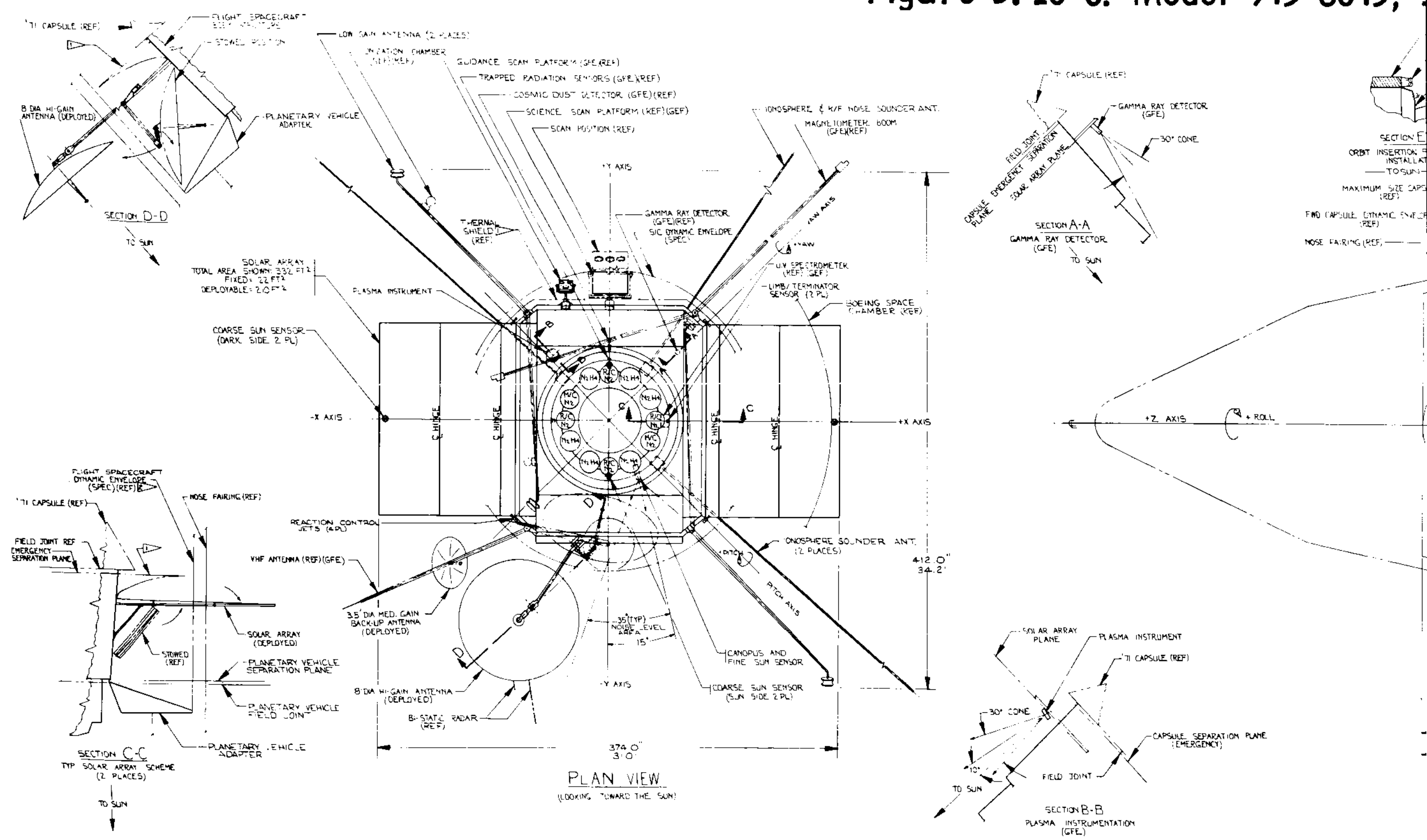
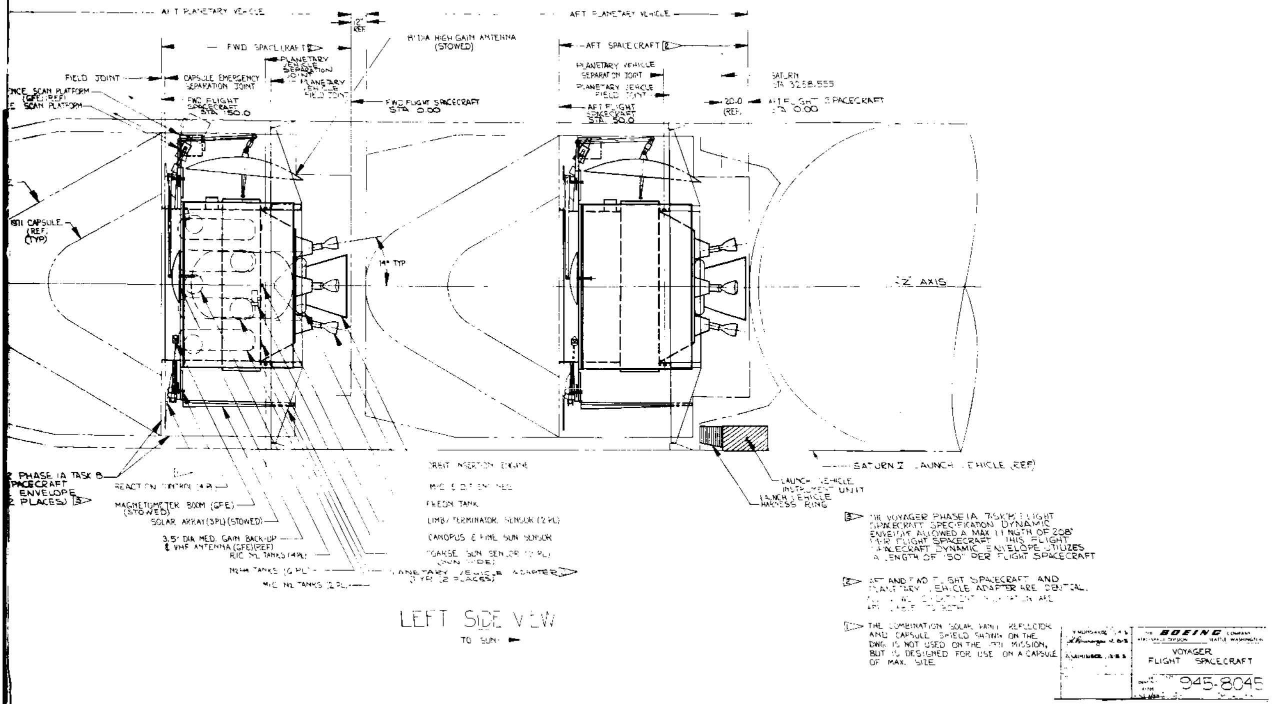
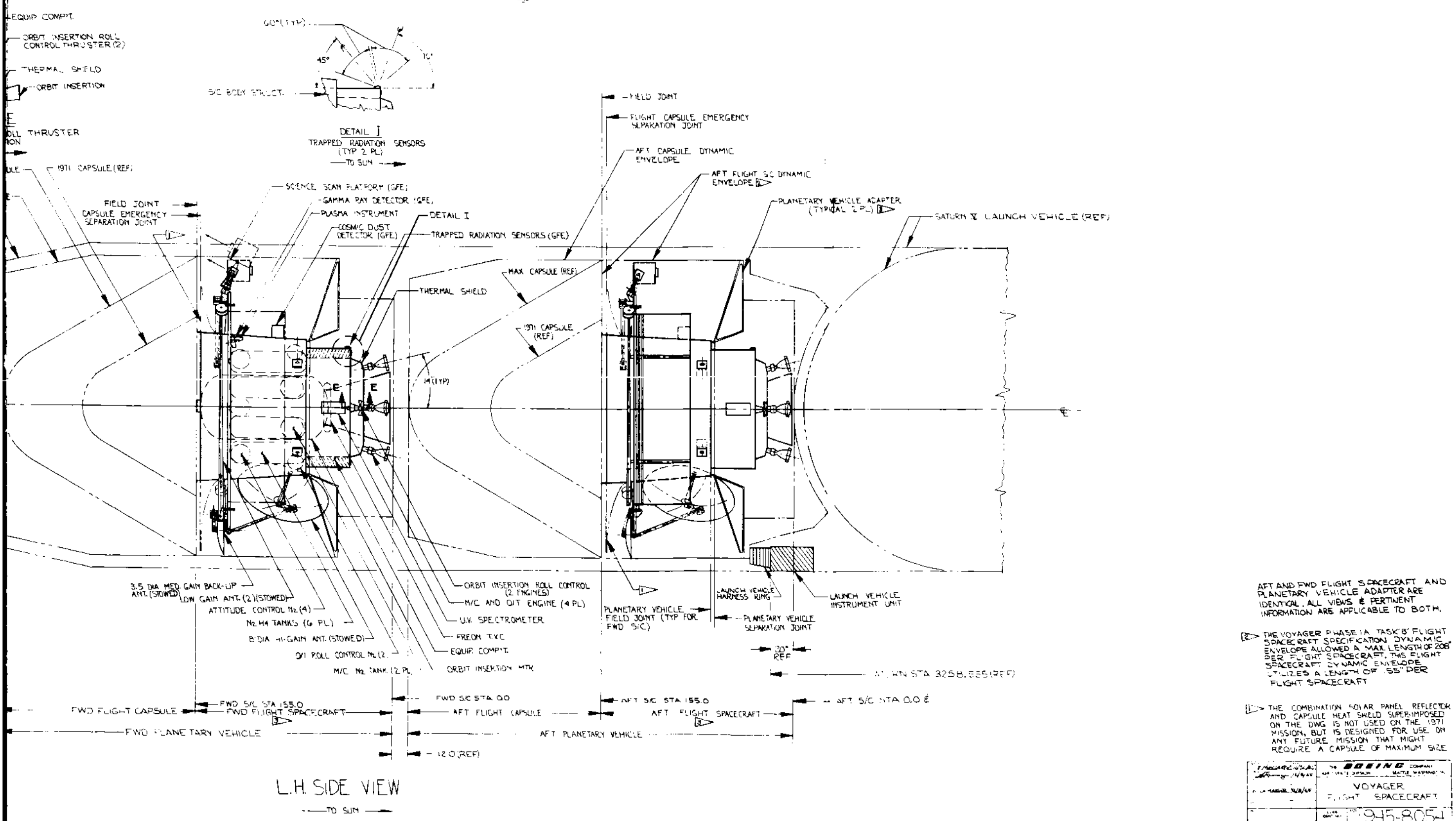


Figure 3.10-7: Model 945-8054, 2



st Alternate Spacecraft Configuration



nd Alternate Spacecraft Configuration

Adapter attaches to the spacecraft at the lower mid-body location rather than at the base of the primary structure. The spacecraft weighs approximately 50 pounds more than the preferred design.

The third candidate configuration, Model 945-8033, is quite similar to the preferred design. Major differences between the two configurations are the requirement for a solid-motor exhaust shield to protect solar panels from excess heating by solid motor exhaust plume, the solar-panel shape and the number of deployed panels, and the moment arm of the science-scan-platform deployment mechanism. The spacecraft weighs 10 pounds less than the preferred design.

3.10.3.2 Evaluation

Measures of value of spacecraft configurations used for rating designs against the competing characteristics defined in Section 1.6 are:

1) Probability of Mission Success--Configuration features that tend to enhance reliability of spacecraft functions, including:
    a) Simplicity of design applied to deployments, load paths, installation volumes, and interfaces;
    b) Modularization of independent functional elements of the spacecraft;
    c) Failure modes that allow a high probability of completing other functions needed for mission success;
    d) Provision for use of redundancy in the spacecraft and weight and volume margins available for additional redundancy.
2) Performance of Mission Objectives--Configuration features that tend to increase the capability of spacecraft systems to meet stated mission objectives, including:

a) Mass properties (weight, moments of inertia, and CG) that provide the ability to enlarge launch time and mission duration windows and to establish and maintain low trajectory and attitude dispersions;

b) Minimum misalignment of scanning, pointing, and inertial reference devices because of deflection or mechanical tolerances in spacecraft mounting structure and deployment elements;

c) Provisions for long-term orbital operations. These include adequate margins in expendables, solar-panel area and the control of equipment thermal environment to moderate and uniform levels.

3) Cost Savings--Configuration features that tend to simplify or eliminate costly functions and facility provisions associated with acquisition, assurance, and activation of the spacecraft system, including:

a) Modularization of independent elements of the assembly, which provides minimum sensitivity of production and test schedules to the spacecraft module interface;

b) The use of developed and flight-proven designs, with emphasis on applications of Mariner C arrangement concepts and components;

c) Spacecraft arrangements that minimize cost of interfacing elements of the Voyager system. Effects of spacecraft arrangements of these interfaces include loads, moments, and separation requirements imposed on the launch vehicle and Flight Capsule; receiving, assembly, countdown, and mission operational requirements imposed on the launch and ground facilities; flight-acceptance testing and type-approval testing facility requirements.

4) Contribution to Subsequent Missions--Ability to accommodate increased capsule size and weight and an increase in quantity and variety of scientific subsystem elements with minimum redesign. Key features include:

   a) A structural design that can be simply modified for load increases;

   b) Excess volume for the addition of science equipment and supporting subsystem growth elements;

   c) Ability to increase solar-panel area with minimum redesign;

   d) View-factor requirements insensitive to capsule diameter and solar-panel area.

5) Additional 1971 Capability--Margins of weight and volume that cannot be used efficiently to improve the baseline mission and that allow for growth to additional performance in the 1971 mission.

All four comparison configurations are capable of performing the 1971 mission. The configurations were evaluated and the results are summarized below.

<u>Probability of Mission Success</u>--Configuration 945-8055 rated highest on inherent simplicity and the ability to accommodate subsystem redundancy. Simple deployments, good c.g. control margins, and solar-panel protection of equipment and many science instruments from the solar environment and orbit-insertion exhaust were the primary features of the 945-8055 configuration that led to its high rating.

<u>Performance of Mission Objectives</u>--All four configurations exceeded mission performance requirements. The two with the solar panels on the Sun side of the equipment, 945-8033 and 945-8055, rated slightly

higher on the basis of more uniform temperatures of critical science equipment and mounting structure throughout the mission.

Cost Savings--None of the arrangements anticipates difficulties in development, fabrication, or system integration. The two configurations with the smallest overall width, 945-8033 and 945-8055, are compatible, with solar panel deployed, with existing full-model test facilities (e.g., the Boeing space chamber at Kent, Washington) and, therefore, would be less costly.

Contribution to Subsequent Missions--Configurations with equipment on the Sun side, 945-8045 and 945-8054, tend to be less sensitive to the size of the capsule and whether it is attached or separated.

Additional 1971 Capability--Configuration 945-8055 rates best in this characteristic because of weight and volume margins, CG control, and solar-panel growth capability.

The evaluations are based on study of the detailed characteristics of each configuration. Consideration of all features results in the selection of Model 945-8055 as the preferred configuration. Table 3.10-3 shows the most significant configuration features analyzed in the study, the major variables imposed on each feature, the primary impact of each variation mode on the design, and the evaluation of each configuration on the basis of these effects.

Supplemental data is presented to support evaluation of major effects. Panel thermal characteristics are shown in Figure 3.10-8. The effect of insertion-motor heating on panel location is shown in Figure 3.10-9.

Table 3.10-3: Configuration Variables

CONFIG FEATURE	VARIABLE	MAJOR EFFECTS	945-8033	945-8045	945-8054	945-8055
SOLAR PANELS	FORE AND AFT LOCATION	Panel Heating Due to Insertion Engine or Back Radiation Blockage View Occultation of Equipment and Sensors	Panel Heating (Engine) Canopus Clear	Panel Heating (20' Capsule) Canopus Noise	Panel Heating (20' Capsule) Canopus Clear	No Problem Canopus Clear
	DEPLOYMENT MODE	• Power Loss and View Blockage Due to Failure to Deploy • Complexity Due to Multiple Hinges and Sequences • Effect on Vehicle CG	Good Failure Modes CG Fair	Poor Failure Modes CG Poor	Poor Failure Modes CG Poor	Good Failure Modes CG Good
	PANEL SHAPE	• Structural and Fabrication Complexity • Area Efficiency • Growth	Low Weight No Growth	High Weight Low Growth	High Weight Good Growth	Low Weight Fair Growth
ANTENNA	CLOCK POSITION	• Occultation During Mission	None	None	None	None
	STOWAGE POSITION	• Failure Modes on Canopus View • Separation Effects • Reliability and Tolerance Effects of Deployment and Number of Joints	Double Hinge Canopus Clear 8'x12'Ellipse Maximum	Double Hinge Canopus Noise Penetrates Adapter 8' Diameter Maximum	Single Hinge Canopus Clear 8'x10'Ellipse Maximum	Single Hinge Canopus Clear 8'x12'Ellipse Maximum
EQUIPMENT PACKAGING	LOCATION	• Radiation Views (20 Ft. Capsule)	19 Watts/Ft^2 Average	25 Watts/Ft^2 Average	25 Watts/Ft^2 Average	19 Watts/Ft^2 Average
	DIAMETER	• Volume for Equipment and CG Adjustment • Area for Thermal Radiators	V = 38 Ft^3 A = 57 Ft^2	V = 38 Ft^3 A = 57 Ft^2	V = 26 Ft^3 A = 45 Ft^2	V = 38 Ft^3 A = 57 Ft^2
PLATFORMS AND SENSORS	LOCATION	• Scan Platform Occultation During Mission • CG Effect of Platform Deployment • Canopus Sensor Spurious Light View Factors	Scan View: Very Good CG: Fair Mounting: Dark Side	Scan View: Adequate CG: Poor Mounting: Sun Side	Scan View: Good CG: Poor Mounting: Sun Side	Scan View: Good CG: Good Mounting: Dark Side
ADAPTER	LOCATION	• Effect on Spacecraft Loads and Moments • Effect on Separation Complexity	Nominal S/C Loads Good Separation	Nominal S/C Loads Antenna Penetrates	Lower S/C Loads Equipment Penetrates	Nominal S/C Loads Good Separation

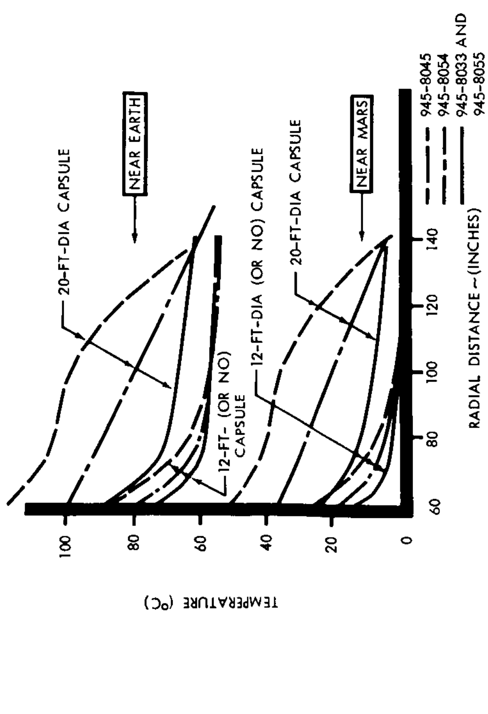


Figure 3.10-8: Solar Panel Thermal Characteristics

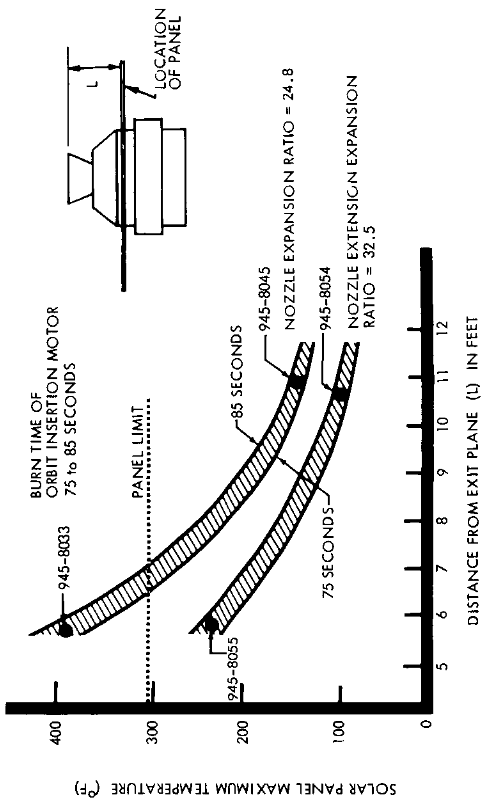


Figure 3.10-9: Solar Panel Heating — Solid Motor Plume

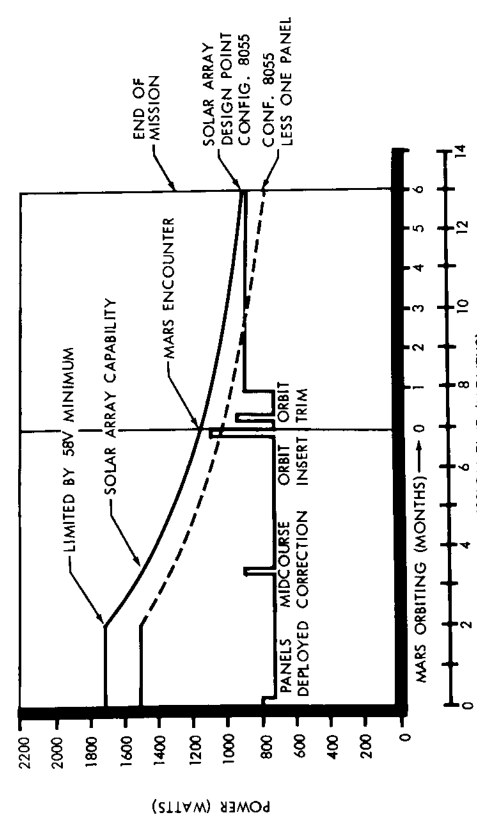


Figure 3.10-10: Power vs. Mission Time

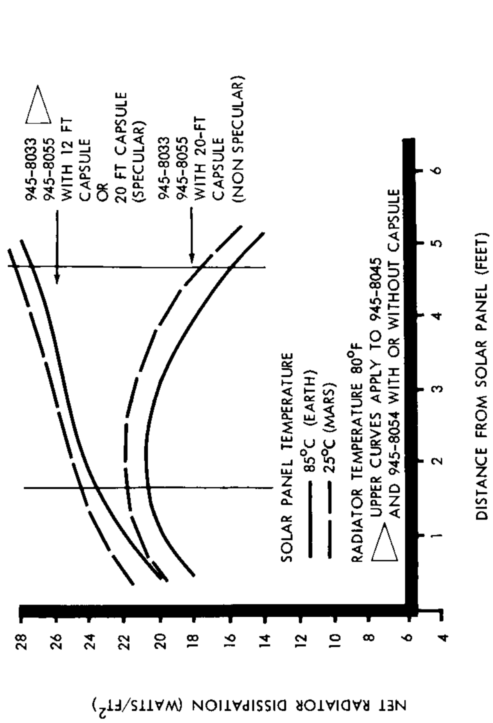


Figure 3.10-11: Equipment Radiator Efficiency

Effects of solar-cell degradation and loss of solar-panels are shown in Figure 3.10-10. Equipment radiator efficiency, as a function of location, is shown of Figure 3.10-11. Important solar-panel design features relevant to the evaluation are summarized as follows:

945-8033 -- Aft position (away from capsule); six folds, 7.1-percent area loss in one deployment failure;

945-8045 -- Forward position; three folds, 24.6-percent loss per failure;

945-8054 -- Forward position; four folds, 15.8-percent loss per failure;

945-8055 -- Aft position; four folds, 10.9-percent loss per failure.

3.11 SELECTION OF PREFERRED SPACECRAFT SYSTEM

This section contains an evaluation of several possible Spacecraft Systems and explains selection of the system to best accomplish the Voyager Program objectives. The evaluation was necessary to ensure inclusion of total system considerations in choosing a Spacecraft System. Three Spacecraft Systems were established based on recommendations of system-level trade studies (Section 3.11.5) and preferred designs of each system part: i.e., propulsion subsystems (Volume C), the Spacecraft Bus subsystems (Section 3.10 and 4.0 of Volume A), the OSE subsystems (Volume B), and trajectories (Sections 3.1 and 3.2 of Volume A). The three systems were compared and the best was chosen. Study conclusions are presented as recommendations with emphasis placed on visibility of the data and the rationale used in arriving at the conclusion. A similar study was performed in Phase IA Task A and was reported in Section 3.10 of Volume A and Section 3.2 of Volume B, D2-82709-1 and -2. It is proposed that these studies be continued throughout Phases IB and II as a means of continual assessment of the Spacecraft System against objectives.

3.11.1 Summary

The Spacecraft Systems selected for final evaluation are:

1) Configuration 945-8055, a modified Wing VI Minuteman second stage solid motor as the orbit-insertion motor with a monopropellant mid-course subsystem sized for the 1975-1977 missions, and an OSE subsystem which is an adaptation of the Mariner OSE. This system will be identified through this section as the "Spacecraft System--Solid."

2) A combination of Spacecraft Configuration 945-8029, a shortened Titan IIIC transtage and the preferred OSE subsystem. This system will be referred to as the "Spacecraft System--T-IIIC."

3) A combination of Spacecraft Configuration 945-8012, a minimum modification to the LEM descent propulsion system and the preferred OSE. This system will be referred to as the "Spacecraft System--LEM." "

The propulsion subsystem variations caused the most significant differences at the Spacecraft System level. The OSE was held constant because the effect of its variations on the Spacecraft System is less significant than the other subsystems. The type of trans-Mars trajectories and the orbit around Mars also were held constant to form a basis for the comparison of the Spacecraft Systems. The trajectories used provide an arrival velocity between 2.82 and 4.5 kilometers per second; the orbit around Mars has a 1000-kilometer periapsis and a period of 13.8 hours.

The three Spacecraft Systems were evaluated by assessing each system by evaluation criteria that measure their capability with respect to the five competing characteristics of the "Preliminary Mission Description", October 15, 1965 (see Section 3.11.2). Table 3.11-1 lists the criteria used, summarizes assessment results, and compares the assessments to system requirements or goals. The results are for Spacecraft System weights as shown in Section 1.5 (Spacecraft Bus, 2100 pounds; science, 400 pounds; propulsion, 15,000 pounds; adapter, 1500 pounds; and a 2000-pound capsule). The effect of the launch vehicle and Science Subsystem are also shown in the probability of success assessments to allow comparison with mission goals. Spacecraft System--Solid was selected as the preferred system because of its low technical risk, and the most weight available for reallocation to further enhance mission success and the lower priority criteria.

Figure 3.11-1 compares estimated probabilities of mission success (P_s) for the preferred system with the goals stated in Section 2.1.2. The curves show two interpretations of mission success; each curve considers both Planetary Vehicles. The top curve is the probability of mission success based on the assumption that science payload failures have no effect on success. It shows that all P_s goals can be met on this basis. The lower curve is based upon the assumption that at least one of each orbital scientific experiment must survive in one of the two Planetary Vehicles for mission success. This curve shows that the probability of obtaining data is high and that the goal for 30 days in orbit can be met.

The weights allocated by JPL are higher than required for the 1971 mission for the propulsion subsystem and the adapter, but are marginal for the Spacecraft Bus. In studying the effect of reallocating the weight, it

Table 3.11-1: Evaluation Summary

Criteria	Goals	Spacecraft System		
		Solid	T-III	LEM
<u>Probability of Success,</u>				
Successful Mission Operation				
1) Launch and Injection	0.85	←	0.87	→
2) Separation and Transit	0.84	←	0.86	→
3) Place in Mars Orbit and Correct (1 of 2 Vehicles)	0.78	←	0.85	→
4) Capsule Separation and 30-day Orbit Operation (1 of 2 Vehicles)	0.65	←	0.83	→
Technical Risk	Minimize	Low	Moderate	Moderate
Schedule Risk	Minimize	←	Very Low	→
Weight Available for Possible Reallocation (lb)	Maximize	810 ▷2	710	700
<u>Performance of Mission Objectives</u>				
Maneuver Accuracy ▷1 (Orbit Period Error Subsequent to Orbit Trim-Seconds) ▷2	Minimize	50	252	68
Potential ΔV Margin (m/sec)	Maximize	140 ▷2	260	250
Orbit Versatility	Maximize	←	Very Good	→
<u>Cost Savings</u>	Maximize	←	Not Evaluated	→
<u>Contributions to Subsequent Missions</u>	Maximize	←	About Equal	→
<u>Additional 1971 Mission Capability</u>				
Additional or Redundant Experiments	Maximize	Best	Good	Good
Alternate Mission Capability	Maximize	Good	Very Good	Very Good

▷1 See Section 3.1.2.2 of Volume C for orbit conditions

▷2 Based on trading monopropellant only (would be approx. 1000 lbs. and 350 m/sec if both solid and monopropellant were traded)

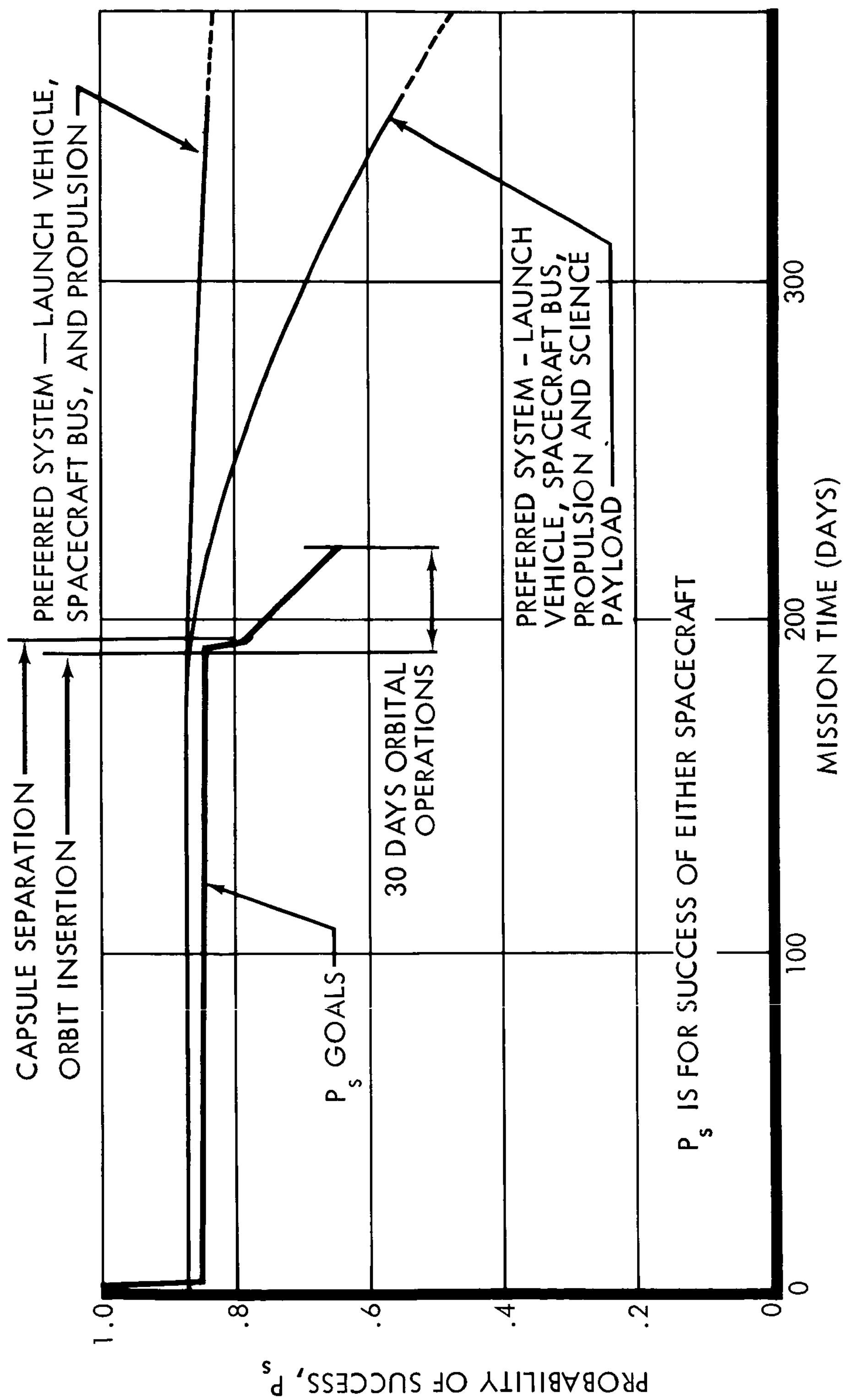


Figure 3.11-1: Probability of Success — Preferred System

was found that all Spacecraft Systems are improved but that their order of preference does not change.

3.11.2 Evaluation Criteria

The evaluation criteria used in Table 3.11-1 are defined in the following paragraphs.

Probability of Success--This is the probability that the system will operate within allowable tolerances and obtain the data required to accomplish the mission objectives. Measures of probability of success are:

1) Probability that the Spacecraft System and Science Subsystem will operate for a specified time and obtain the desired data;
2) Probability of a partial success, i.e., obtain part of the desired data;
3) Technical risk--a measure of deviation from the state of the art;
4) Schedule risk--a measure of the risk of not meeting the launch date if all systems are within the state of the art;
5) Weight available for possible reallocation--the Spacecraft System weight that can advantageously be reallocated using an increase in the probability of mission success as the highest priority.

Performance of Mission Objectives--Assessment is made on the basis of how well each system can accomplish the mission objectives assuming a 1.0 probability of success, e.g., orbit accuracies and the range of possible orbits around Mars.

Cost Savings--Assessment is made on the basis of the differences in total costs for the total Voyager 1971 mission.

Contributions to Subsequent Missions--Assessment is made on the ability of the basic system to perform subsequent missions in terms of sufficient excess propulsion capability.

Additional 1971 Mission Capability--This term is the capability of the system beyond that required to accomplish minimum objectives of the 1971 mission, measured in terms of additional or redundant experiments and alternate mission capabilities.

3.11.3 Selection Rationale

This section describes the assessment of each Spacecraft System against each criterion and the process used to select the preferred Spacecraft System.

3.11.3.1 Assessment of Spacecraft Systems against Evaluation Criteria

Probability of Success--The probability of success for the three Spacecraft Systems is nearly the same in all areas except technical risk and the weight available for reallocation. The Spacecraft System--Solid is rated as having the lowest risk. The hydrazine monopropellant system is of the same basic design concept as the Mariner propulsion system. Because the solid motor orbit-insertion propulsion system is hermetically sealed, the propellant is maintained in an environment similar to that in Earth storage. Since motors of this type have demonstrated high storage and operational reliability, it is considered to have low technical risk. The systems using the Titan IIIC transtage and LEM propulsion systems are rated as having higher technical risk. Both systems require several modifications to adapt them to the Voyager mission, and the effect of the space environment on long term storage of bipropellant propulsion systems is unknown. In addition, propellant leakage may result in residue that could cause component failure.

For evaluating probability of success, it is assumed that weight available from the unused propulsion subsystem allocation could be reallocated and used on the Spacecraft Bus to further maximize P_s. Therefore, the system with the highest weight margin was rated best. The values shown in Figure 3.11-1 were calculated for the nominal orbit around Mars. When using orbits approaching minimum and maximum periods (3 and 24 hours per Figure 3.1-26), certain redundant electrical components must be turned off after 30 days in orbit (see Section 4.1.1.5). Losing this redundancy would cause P_s to be degraded from 0.83 to 0.81.

Performance of Mission Objectives--For this criterion, the Spacecraft System--Solid is about equal to Spacecraft System--LEM. The maneuver accuracy of the Spacecraft System--Solid is best followed by that of the Spacecraft System--LEM. Potential ΔV margin is best for Spacecraft System--T-IIIC closely followed by Spacecraft System--LEM. Orbit versatility is a measure of the ability of the spacecraft to achieve certain orbits around Mars. The comparison is based on different combinations of launch and arrival dates that can be achieved for each system when considering minimum ΔV, effect of orbit period, and orientation. All systems are about equal with respect to this criterion, as explained in Volume C, D2-82709-8.

Cost Savings--Due to the lack of firm cost data, cost differences could not be used in the evaluation.

Contributions to Subsequent Missions--The criterion used is the growth potential of the vehicles in terms of their capability in performing subsequent missions. All systems are designed for the most difficult missions; thus, all are rated equal.

Additional 1971 Capability--On the basis of potential weight that could be reallocated to increase redundancy in experimental equipment or to add experiments, the Space System--Solid is best. On the basis of alternate mission capability, it is rated lowest due to a lower ΔV margin and the lesser flexibility of the solid insertion motor.

3.11.3.2 Selection of Preferred Spacecraft System

The preferred Spacecraft System was selected using two procedures. The first considers the ratings of each system against each criterion, with consideration for the priority of the competing characteristics. With this method, the Spacecraft System--Solid is best in probability of success; for performance of mission objectives, contributions to subsequent missions, and additional 1971 mission capability, all systems are about equal.

The second procedure is an application of Fishburn's independence-in-utility theory (Reference 5) in which relative values and an additive rating procedure are used.

Reference 5: Fishburn, P.C., "Independence in Utility Theory with Whole Product Sets," CORS/ORSA Joint Conference, Montreal, Canada, May 27, 1964. Also used by Boeing in the study "System Criteria for Launch Vehicle Systems, " NAS8-11429, May 1965.

This method involves:

1) Assignment of relative or weighted values to each of the competing characteristics and their subelements;
2) Conversion of the individual assessments of Table 3.11-1 to values, giving the full value to the best system, zero to the least promising, and an intermediate value to those between;
3) Determination of total assessment: i.e., the sum of the individual values for each Spacecraft System;
4) Identification of the preferred system;
5) Determination of the sensitivity of the selection to a change in assessments and relative values.

Table 3.11-2 shows results of applying this system to the data of Table 3.11-1. Sensitivity studies showed that the selection is not sensitive to the weighted values assigned unless the priority of the competing characteristics is changed. This evaluation confirms Spacecraft System--Solid as the preferred system.

3.11.4 Reallocation of Spacecraft System Weights

The evaluation of Section 3.11.1 uses the weights available for possible reallocation as one of the evaluation criteria. This section presents the results of a study to determine potential uses for the available weight and shows an example of reallocation for each Spacecraft System.

3.11.4.1 Summary

Table 3.11-3 shows the reallocations made and the effect of these reallocations on the two evaluation criteria that are most affected, P_s and performance of mission objectives. It is shown that P_s can be materially

Table 3.11-2: SELECTION SUMMARY

Criteria	Maximum Values	Spacecraft System Values		
		Solid	T-III	LEM
Probability of Success	417			
Successful Operation	217	217	217	217
Technical Risk	100	100	0	0
Schedule Risk	50	50	50	50
Reallocated Weight	50	50	0	0
Performance of Mission Objectives	236			
Maneuver Accuracy	76	76	0	68
Potential ΔV Margin	80	0	80	73
Orbit Versatility	80	80	80	80
Cost Savings	180	180	180	180
Contributions to Subsequent Missions	111	111	111	111
Additional 1971 Mission Capability	56			
Additional or Redundant Experiments	28	28	0	0
Alternate Mission Capability	28	0	28	28
Total Assessments		892	746	807

improved for all Spacecraft Systems, that the ΔV margin is decreased for all systems, and that the Spacecraft System--Solid and LEM can obtain improved performance when using 3- and 24-hour orbits.

It is concluded that all of the Spacecraft Systems are improved by the reallocation, but that their relative ratings are not changed.

Table 3.11-3: REALLOCATION OF WEIGHT

WEIGHT ALLOCATIONS					APPLIED TO SPACECRAFT SYSTEM	P_s 6 MO. IN ORBIT W/SCIENCE	PERFORMANCE OF OBJECTIVES	
ALLOC. PER	BUS	SCIENCE	ADAPTER	PROPULSION			ΔV MARGIN M/SEC	IMPROVE 3 TO 24 HR ORBIT PERF.
Sect. 1.5	2100	400	1500	15,000	Solid T-IIIC LEM	.49 .48 .51	130 173 115	no no no
New	2355	920	850	14,875	Solid	.70	11	yes
New	2313	897	880	14,910	T-IIIC	.69	0	no
New	2255	920	600	15,225	LEM	.70	16	yes

3.11.4.2 Weight Available for Reallocation

Figure 3.11-2 indicates for the three alternate systems the weights that are available for reallocation while meeting the Planetary Vehicle weight of 21,000 pounds, a capsule weight of 2,000 pounds, and an orbit-insertion ΔV of 2.2 kilometers per second. The figure indicates that for the Spacecraft System--Solid, the bus weight could be increased by 810 pounds (from the allocated 2,100 pounds) to a maximum of 2,910 pounds. This increase necessitates an increase in propulsion weight of 1,160 pounds from the present weight of 13,680 pounds to 14,840 pounds to retain at least a

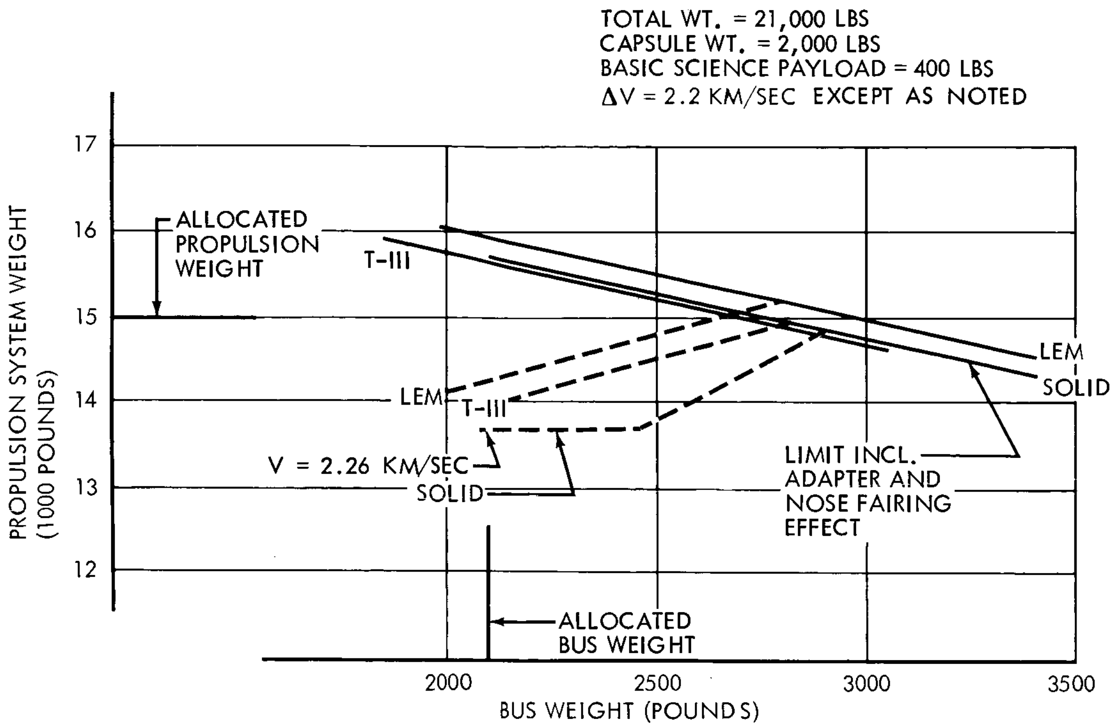


Figure 3.11-2: Weight Allocation

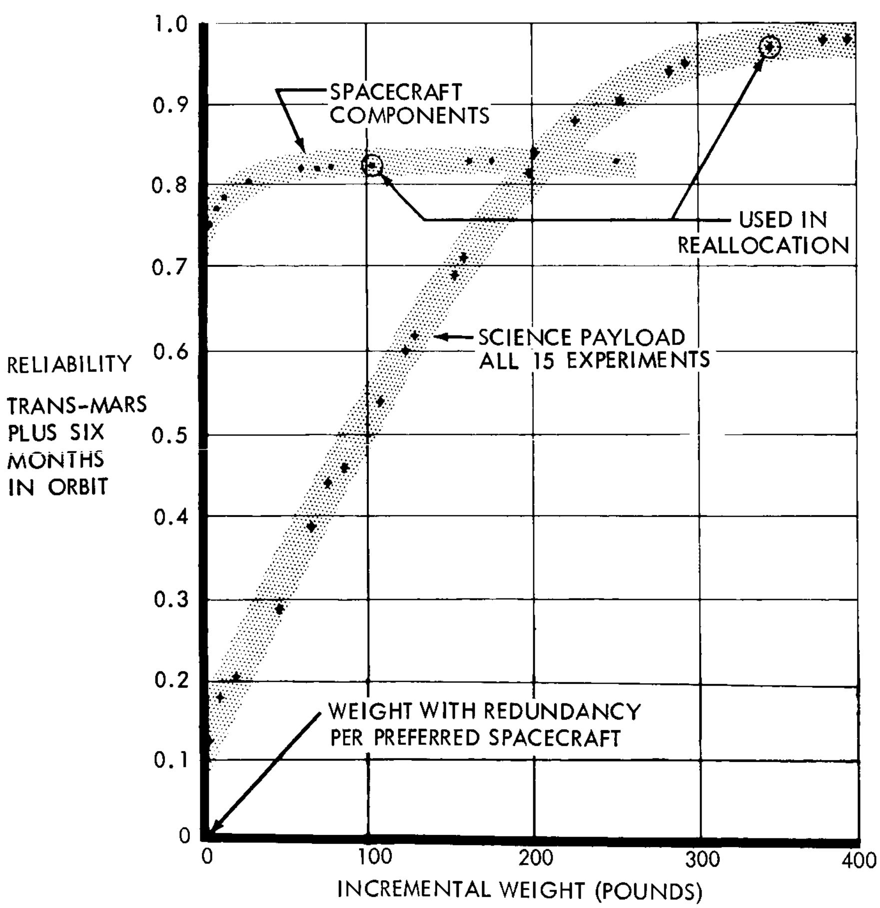


Figure 3.11-3: Sequence of Optimal Weight Versus Reliability

2.2-kilometer-per-second ΔV capability. Alternately, the weight margin could all be allocated to the propulsion; in that case, a maximum propulsion weight of 15,650 pounds is possible.

3.11.4.3 Weight Reallocation

Reallocation of weight was considered in the order of priorities of the competing characteristics. The following paragraphs summarize the items considered.

Probability of Mission Success (P_S)--Additional redundancy in the Spacecraft Bus and the Science Subsystem was studied. This was studied using a method developed by Dr. Frank Proschan of the Boeing Scientific Research Laboratories (Reference 6), in which the components selected result in the maximum increase in reliability per unit of weight added. Figure 3.11-3 shows the improvement possible over the preferred configuration as Spacecraft Bus and science payload components are made redundant and shows that the addition of more than 103 pounds of Spacecraft Bus components or 346 pounds of Science Subsystem equipment will result in little further improvement. In making reallocations, another 50 and 174 pounds, respectively, must be added to provide for required structure and electrical power.

Performance of Mission Objectives--Two items were considered. One was the use of additional propellant for midcourse corrections and orbit-trim maneuvers (max ΔV). The other was the addition of 102 pounds to the electric power subsystem to avoid turning off certain redundant electrical components prior to the end of the mission when in orbits approaching expected minimum and maximum periods (3 hours and 24 hours).

Reference 6: Proschan, Frank, and Barlow, R.E., Mathematical Theory of Reliability, John Wiley and Sons, 1965.

Contribution to Subsequent Mission--Only the addition of 500 pounds to the spacecraft adapter (Solid and T-IIIC Spacecraft Systems) was considered. This would allow a design that could be used without change for the 1975 to 1977 missions.

In reallocating the weights as shown in Table 3.11-3, only the items under P_s and performance of mission objectives were used.

3.11.5 System Concept Trade Studies

Trade studies were conducted to determine the optimum method for meeting system and subsystem requirements and to provide a basis for selecting the preferred design. The process followed was to:

1) Identify the required trade studies from reviews of the Task A effort and the Task B statement of work;
2) Sequence the trades so that independent trade studies would be conducted prior to dependent ones;
3) Establish the evaluation criteria, related to the competing characteristics, for performing the trade;
4) Conduct the trade and select a recommended approach;
5) Verify that all pertinent trades were performed;
6) Use the trade study results in the design.

The trade studies reported in this document and in Volumes B and C are based on the mission description and first-level functional flows. As the program proceeds, further trades will be performed, including those at lower functional levels, to achieve optimum system performance and compatibility.

The system concept trade studies conducted during Task B (except for propulsion and mission trades) are summarized in Table 3.11-4. The competing characteristics were of primary consideration in trade study decisions. Detailed discussions of each of the listed trades are found in the referenced subsections of Volume A, or if noted in Volume B, and propulsion trades are discussed in detail in Volume C. Mission trades are summarized in Section 1.7 of this document.

TABLE 3.11-4: SYSTEM TRADE STUDY SUMMARY

TRADE STUDY TITLE	SELECTED ALTERNATIVE	RELATIONSHIP TO COMPETING CHARACTERISTICS					REF DOC SECTION NO.
		PROBABILITY OF SUCCESS	PERFORMANCE OF MISSION OBJECTIVES	COST SAVINGS	CONTRIBUTION TO SUBSEQUENT MISSIONS	ADDITIONAL 1971 MISSION CAPABILITY	
Determine Method and Location of Magnetic Mapping Facility and OSE (System Level)	Earth ambient mapping at both Seattle and KSC	P	N/A	S	N/A	N/A	Vol. B 3.3.4
Transport of Spacecraft from Seattle to KSC	Air transport--spacecraft assembled	P	N/A	S	N/A	N/A	Vol. B 3.3.4
Free Mode Test	Perform test at Seattle (Kent) using artificial light	P	N/A	P	N/A	N/A	Vol. B 3.3.4
Computer and Sequencer (C&S) Subsystem Configuration	Two dependent synchronous C&S subsystems using internal cross checking and data transfer.	P	N/A	N/A	N/A	N/A	4.1.9
Method for Providing Cumulative Time Reference	Three separate registers in C&S memory under stored program control.	P	P	P	N/A	N/A	4.1.9
C&S Design Approach Determination	Special-purpose data processor.	P	N/A	P	N/A	N/A	4.1.9
Solar Panel Deployment Mechanism Deployment Actuator	Clockspring at each hinge line.	P	N/A	N/A	N/A	N/A	4.1.1
Solar Panel Configuration	Corrugated aluminum substrate	P	N/A	P	S	N/A	4.1.1
Guidance and Control Subsystem Gyro Configuration	Three double-axis free-rotor gyros	P	N/A	N/A	N/A	N/A	4.1.2
Guidance and Control Subsystem Planet Sensor Mechanization	Separate pointing platform for guidance sensors	Competing characteristics not involved in selection. Alternative selected on the basis of preserving simplicity of interface. May be re-exercised when details of science payload become available.					4.1.2
Selection of Planetary Vehicle Adapter Structural Arrangement and Material	Semimonocoque (skin and stiffener) design (magnesium)	Competing characteristics not involved in selection. Selection based on weight, accessibility, fabricability.					4.2.1

CODE: P--Selected alternative was rated highest in comparison with other alternatives for this competing characteristic.

S--Selected alternative ranked lower than other alternatives for this competing characteristic.

N/A--Competing characteristic not applicable.

TABLE 3.11-4: SYSTEM TRADE STUDY SUMMARY (CONTINUED)

TRADE STUDY TITLE	SELECTED ALTERNATIVE	RELATIONSHIP TO COMPETING CHARACTERISTICS					REF DOC SECTION NO.
		PROBABILITY OF SUCCESS	PERFORMANCE OF MISSION OBJECTIVES	COST SAVINGS	CONTRIBUTION TO SUBSEQUENT MISSIONS	ADDITIONAL 1971 MISSION CAPABILITY	
Planetary Vehicle Adapter Location	Located to provide support at the base of the Planetary Vehicle.	Competing characteristics not involved in selection. Selection based on economy of space, minimum weight in orbit.					4.2.1
Thermal Isolation of the Propulsion Module	Propulsion model thermally isolated from Spacecraft Bus.	P	N/A	N/A	N/A	P	4.1.12
Solid Motor Exhaust Thermal Effects Reduction	Changes in motor nozzle expansion rates and length.	P	N/A	N/A	P	N/A	4.3.5
Spacecraft Primary Structure Selection	Semimonocoque (sheet and stringer construction using eight partial longerons).	Competing characteristics not involved in selection. Selection based on weight, access to equipment, and relative complexity.					4.1.10
Integration of Electronic Packaging with Structure	Electronic packages located on exterior of structural skin. Packages do not carry primary structural loads.	P	N/A	N/A	N/A	N/A	4.1.10
Selection of Propulsion Subsystem Support Structure	Primary truss and shell structure located above solid motor attach point.	Competing characteristics not involved in selection. Selection based on access to motor for installation and inspection and to attitude control group.					4.3.4
Selection of Antenna Drive Method	Harmonic drive with magnetic detent stepping motor and magnetic noncontacting-type encoder.	P	N/A	N/A	N/A	N/A	4.1.2
Reaction Control Subsystem Tankage Redundancy Study	Two separate tankage and regulator systems.	P	N/A	P	N/A	N/A	4.1.2
Momentum Exchange Device Study	Pure reaction control system.	P	N/A	N/A	N/A	N/A	4.1.2
Reaction Control Subsystem--Cold Gas Propellant	Use of cold nitrogen gas.	P	N/A	P	N/A	N/A	4.1.2
Reaction Control Subsystem Thrust Level Trades	Single level of thrust for use before and after capsule separation.	P	N/A	P	N/A	N/A	4.1.2

CODE: P--Selected alternative was rated highest in comparison with other alternatives for this competing characteristic.

S--Selected alternative ranked lower than any other alternatives for this competing characteristic.

N/A--Competing characteristic not applicable.

TABLE 3.11-4: SYSTEM TRADE STUDY SUMMARY (CONTINUED)

TRADE STUDY TITLE	SELECTED ALTERNATIVE	RELATIONSHIP TO COMPETING CHARACTERISTICS					REF DOC SECTION NO.
		PROBABILITY OF SUCCESS	PERFORMANCE OF MISSION OBJECTIVES	COST SAVINGS	CONTRIBUTION TO SUBSEQUENT MISSIONS	ADDITIONAL 1971 MISSION CAPABILITY	
Capsule Canister Configuration--Effect on Equipment Packages and Solar Panels	Canister configuration incorporates a specular 75-degree half-angle cone at base.	N/A	P	P	P	N/A	4.1.12
Equipment Package Coupling to Structure	Conduction from electronic modules to radiater plate, other modules, and structures.	P	N/A	N/A	N/A	N/A	4.1.12
Pyrotechnic Drive Circuit Design	Two transistor drives in series for critical circuits.	P	N/A	N/A	N/A	N/A	4.1.11
	Single silicon controlled rectifier for noncritical circuits.	P	N/A	N/A	N/A	N/A	4.1.11
Pyrotechnic Subsystem Redundancy Study	Command--C&S cooperative redundancy--two inputs, two EED's, and two drivers for critical circuits.	P	N/A	N/A	N/A	N/A	4.1.11
	Command--C&S cooperative redundancy--single input for noncritical circuits.	P	N/A	N/A	N/A	N/A	4.1.11
Telecommunications Configuration Study	Use of Biorthogonal Encoding for Planetary Science Data: Antenna--6.5 feet, two axes.	P	P	S	P	P	4.1.3
Data Storage Subsystem, Photo Imagery No. 1 and 2, 1.2×10^8 Bit Capacity Magnetic Tape Recorder	Continuous parallel record/reproduce.	N/A	P	N/A	N/A	N/A	4.1.4

CODE: P--Selected alternative was rated highest in comparison with other alternatives for this competing characteristic.

S--Selected alternative ranked lower than other alternatives for this competing characteristic.

N/A--Competing characteristic not applicable.

3.12 PLANETARY QUARANTINE

This section describes the techniques employed for complying with the planetary quarantine probability apportionments listed in Section 2.5. The following procedures are proposed to ensure that the probability allocation (shown in parentheses) for each contaminating event is not exceeded.

Fairing of Saturn IVB Stage Nose (0.2×10^{-5})--The S-IVB stage will be retrofired after Planetary Vehicle separation to place it on a Mars nonimpact trajectory. The forward nose fairing will be released prior to trans-Mars injection and the S-IVB/aft nose fairing on a Mars nonimpact trajectory.

Capsule Canister Impact (0.1×10^{-5})--Capsule canisters (or biobarriers) will be separated while the Planetary Vehicle is on a nonimpact trajectory.

Flight Capsule Contributions (2.3×10^{-5})--The Flight Capsule will be sterilized, and procedures will be developed to ensure that sterility is maintained throughout test and integration.

Flight Spacecraft Accidental Impact (1.0×10^{-5})--The aim points will be biased for each spacecraft. Insertion command will not be given unless the resulting orbit satisfies this constraint.

Propulsion Subsystem Exhaust Products (0.4×10^{-5})--Although preliminary information indicates that engine sterilization requirements may be obviated, it appears prudent at this time to continue to plan for the sterilization of some of these systems.

Orbit-Insertion Subsystem (0.2×10^{-5})--The solid propellant and associated components of the orbit-insertion engine will not require sterilization or treatment to satisfy the planetary quarantine constraint. The TVC fluids (freon and nitrogen) will be sterilized by filtration and aseptically loaded. Tanks and associated plumbing for these systems will be decontaminated prior to assembly and maintained in this condition during assembly, checkout, and launch operations. The decontamination procedure may be accomplished either by: (1) ethylene oxide treatment or (2) heat sterilization, according to NASA/JPL Specification XSO-30275-TST-A. Either procedure will be acceptable to satisfy the planetary quarantine constraint.

Orbit-Trim Subsystem (0.2×10^{-5})--Neither the propellant (hydrazine) nor the orbit-trim subsystem will require sterilization or decontamination. Hydrazine is sporicidal, and components in contact with the propellant will be sterilized.

Reaction Control Subsystem Exhaust Products (0.1×10^{-5})--Internal surfaces of the reaction control subsystem hardware (tanks, valves, nozzles, tubing, etc.) will be decontaminated prior to assembly and maintained in this condition throughout assembly, checkout, and launch. Either ETO treatment or heat sterilization treatment, according to NASA/JPL Specification XSO-30275-TST-A, will be suitable for accomplishing decontamination.

Spacecraft Meteoroid Impact Ejecta (1.0×10^{-5})--Analysis indicates that biological contamination of Mars resulting from ejecta from the spacecraft external surfaces and appendages being subjected to meteoroid impact may not be a problem. Recent studies of the bactericidal effects of ultraviolet radiation and the emissivity of spores indicate that the exposure to solar radiation and heat pulse encountered during entry through the Mars atmosphere might reduce the viable organism survival probability to a level approximating the allocation. However, validation of the conclusion that decontamination of external surfaces is not required must be based on further tests conducted during Phase IB.

3.13 VOYAGER FLIGHT-EQUIPMENT CLEANLINESS

The approach to ensure flight-hardware cleanliness is concerned with avoiding contamination that could adversely affect the mission. Two types of contamination, particulate and biological, are of special concern. Controls on personnel, processes, and packaging during assembly and test will be exercised to ensure that cleanliness levels prior to encapsulation in the nose fairing will meet the requirement of no particles larger than 4 mils. Investigation to date indicates there will be problems in meeting this requirement, with reasonable cost, for internal surfaces of equipment. Further work will be done in this area. The following procedures will be used.

1) After a trial assembly of all mechanical interfaces, the Flight Spacecraft equipment will be disassembled to the lowest practical level, cleaned, and reassembled in a Class 100,000 laminar downflow facility. If components received from another facility are found to be externally contaminated, they will be cleaned to meet the conditions of clean-room assembly. All subcontractor components intended for direct clean-room assembly will be required to be assembled in Federal Standard 209, Class 100,000 rooms, and to meet specified particulate-contamination requirements. In the case of autopilot gyros and similar equipment, a Class 100 bench environment (Fed. Std. 209) will be required during assembly.

2) The liquid and gaseous elements of the reaction-control and propulsion subsystems, and associated components of these subsystems, will be internally precleaned and then assembled in a Class 100 environment. Procedures for cleaning, inspecting, packaging, and handling of the subsystems and their components have been established in the Lunar Orbiter Program. Failure to meet the standards will result in disassembly and reprocessing.

3) All remaining Flight Spacecraft components will be cleaned, inspected, packaged, and shipped in accordance with existing Boeing procedures established in the Lunar Orbiter Program.

3.14 MAGNETICS

This section discusses the Flight Spacecraft magnetic field in relation to requirements and describes the magnetic-control plan and testing facility.

3.14.1 Magnetic-Control Plan

Section 2.2.3 lists the requirements on the Flight Spacecraft magnetic field. The magnetic-field requirements will be met by:

1) Specifying nonmagnetic materials;
2) Minimizing, through design, the contributions of electrical components;
3) Subsystem testing for monitoring compliance to requirements using the analytic model;
4) Flight Spacecraft magnetic mapping to demonstrate compliance.
5) Maintain records and obtain JPL approval for use of magnetic materials.

3.14.1.1 Materials

Structural and Mechanical--Selection of materials for mechanical and structural application will consider magnetic permeability, magnetic remanence, Curie temperature, and geometry. Materials can be separated into three categories from magnetic to nonmagnetic:

Category I: Nonmagnetic; permeability < 1.01; remanence; zero;

Category II: Magnetic; permeability > 1.1; remanence; very high;

Category III: Slightly magnetic; permeability $> 1.01, < 1.1$; remanence; slight.

Category I materials may be used without constraint and will fulfill the requirements of most of the vehicle structure.

Category II materials will be restricted in usage to applications where no less magnetic material will perform the desired function. The requirements for usage of these materials will be justified in detail and specific project authorization obtained.

Category III materials usage will be minimized because properties of materials in this category vary from sample to sample and with processing history. The use of these materials cannot be eliminated because existing qualified parts and components of systems, such as propulsion and attitude control, are fabricated from these materials.

Electrical Components--The magnetic field of typical electrical components is due to soft and hard permanent fields (Fields 1 and 2 on Table 3.14-1), induced fields (Field 3), and stray electromagnetic fields (Field 4).

Table 3.14-1: MAGNETIC FIELDS OF TYPICAL ELECTRICAL COMPONENTS

Typical Component	Possible Field 1	2	3	4	Typical Field At 3 feet (Gamma)
Capacitor	X		X	X	10^{-2}
Transformer	X		X	X	
Resistors	X		X	X	10^{-3}
Traveling-Wave Tubes	X	X	X	X	14 to 26
Inductors	X		X	X	
Motors (with Shield)	X	X	X	X	1
Solenoid Actuators	X		X	X	1 to 5
Power Supplies	X		X	X	3 to 10
Cordwood Circuitry	X		X	X	1
Transistors (five per pkg)	X		X	X	2×10^{-3}
Television	X		X	X	
RF Circulator Switches	X	X	X	X	10
Lead Wires	X		X	X	
Connectors	X		X		10^{-2}
Isolators	X	X	X		10

The externally induced field (Field 3) is generally small for properly controlled conditions of Field 1, 2, and 4.

Soft permanent fields and stray electromagnetic fields will be of primary concern because they are large and contribute significantly to total magnetic-field instability.

Effects of hard permanent fields will be minimized by designing for the most efficient magnetic circuit, by confinement where required, and by using magnetic shields. Sources with the greatest dipole moments are placed as remote from the magnetometer as possible. The effect of these fields on stability will be minimized by limiting use of ferromagnetic materials.

The effect of stray electromagnetic fields will be reduced by proper design of the electronic layout and the power-transfer system, elimination of ground loops, and compensation of each offending magnetic loop.

The area of any current-carrying loop will be minimized. Tightly twisted wires, individual ground and signal-return wires, and coaxial cables will be used. Components with relatively large electromagnetic fields will be designed to operate in a back-to-back field-canceling mode.

Components susceptible to perming by stray electromagnetic fields will affect the total magnetic-field stability. Stray fields will induce a perm into these components. The design of these components will include selection and control of materials that will minimize this effect.

3.14.1.2 Analytic Approach

An analytic approach that will predict the magnetic field of the Flight Spacecraft at the magnetometer location was studied. The approach outlined was based on the analytic methods reported in a Texas Instrument Incorporated study (Reference 7). The purpose of this prediction technique was to provide guidelines for the geometry, location, and composition of the magnetic components of the Flight Spacecraft to meet the magnetic requirements at the magnetometer.

An analytic model was developed during the study. It was determined that the analytic model could be used to estimate the effect of magnetometer boom length on material selection. It is recognized that use of the model early in the program, before detail design is complete, would require input-data assumptions to be made. Using conservative assumptions, the analysis can be used to determine allowable materials as a

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Reference 7: Green, Arthus W., Jr., and Burch, Jack J., "A Stochastic Approach to the Problem of Allowable Magnetic Moments in a Space Vehicle," Texas Instruments Incorporated, Dallas, Texas.

function of boom length and the effect of various boom lengths on costs and Flight Spacecraft performance.

3.14.1.3 Magnetic-Testing Facility

The JPL magnetic requirements will be met by using the system, component, and assembly test facility, and the magnetic-material test facility.

The system, component, and assembly test facility will be capable of mapping and deperming the Flight Spacecraft or the Planetary Vehicle. Tests on the PTM will help determine potential problems.

A magnetic-material test facility will be used to support magnetics research, design investigations, and quality control on materials, parts, and subassemblies. This facility will allow supplemental activity to the work being done at JPL in an effort to ensure that problems will not appear at the system level.

3.15 SPACE-RADIATION EFFECTS ON VOYAGER SYSTEM

This section is concerned with the radiation effects on components, assemblies, and subsystems due to the predicted space-radiation environment for the Voyager mission.

The radiation effects likely to be most important to the Voyager are surface and ionization effects rather than bulk damage to electronics. Much data exists on basic radiation response of electronics and materials intended for Voyager (Table 3.15-1), and more data on specific devices and materials, especially surface phenomena, is being obtained.

3.15.1 Semiconductor-Device Response

Ionization from charged particles may cause temporary loss of transistor gain, increased leakage current, and reduction in breakdown voltage. Special surface treatments and manufacturing techniques can reduce these effects. Screening and selection of devices, as well as the above techniques, will result in devices that meet Voyager environment requirements.

Solar-cell assemblies will be affected by both the solar wind (low energy) proton and by solar ultraviolet radiation. The NP silicon cells selected for Voyager are more radiation resistant than PN. Experimental data indicates that quartz covers for solar cells will not be degraded by the predicted fluxes.

A detailed evaluation of electronic components, including safety margins and ratios of damage threshold levels to Voyager environment levels, has been accomplished (Table 3.15-1).

3.15.2 Voyager-System Materials

Thermal-control-coating performance will depend on the combined effects of particulate and electromagnetic radiation on the absorptance and emittance properties of the coating. Presently, the combined environmental effects are being studied on selected thermal-control coatings to ensure Voyager thermal-control-system reliability.

Subsystems using semiconductors are the main concern because of their susceptibility to radiation damage. Radiation response of these components will be minimized by selecting low-radiation-response devices and analysis of allowable outputs and inputs from semiconductor circuits to ensure that radiation-induced spurious signals do not cause allowable ranges for proper performance to be exceeded.

Table 3.15-1: THRESHOLD FOR PERMANENT RADIATION DAMAGE FOR TYPICAL ELECTRONICS

Transistor Number	Type	Damage Safety Margin (Interplanetary Mission)*
Silicon Transistors		
2N930	NPN	30
2N708	NPN	120
2N2150	NPN	3.5
2N2331	NPN	50
SP8300 (2N708)	NPN	160
Diodes		
IN649		>100
IN485B		>300
AM 602		>10
IN746A		>500
Capacitors		
CTM	Mylar	>100
VKR	Ceramic	>105
Resistors		
IK	Coated Metal Film	>104
10 R-100K	Glass-Metal Film	>104

*Safety margin is the factor by which the radiation environment must be increased to observe damage.

3.16 DATA SYSTEM DESCRIPTION

The data system for the Voyager 1971 Mars mission is shown schematically in Figure 3.16-1. It acquires, processes, and displays tracking, engineering, and science data, and transmits and verifies spacecraft commands.

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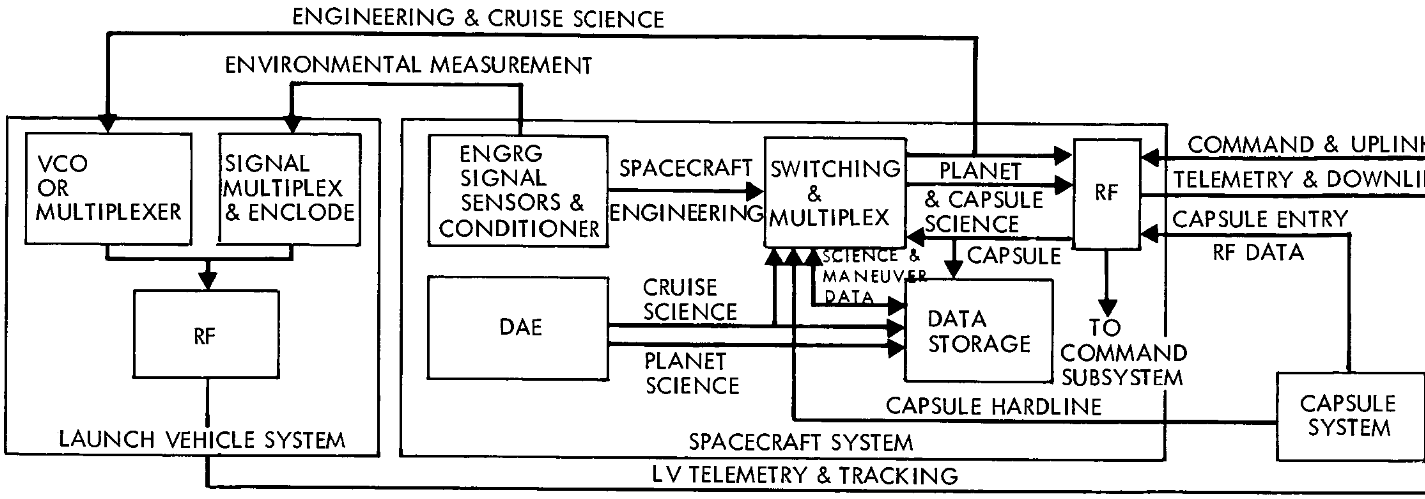
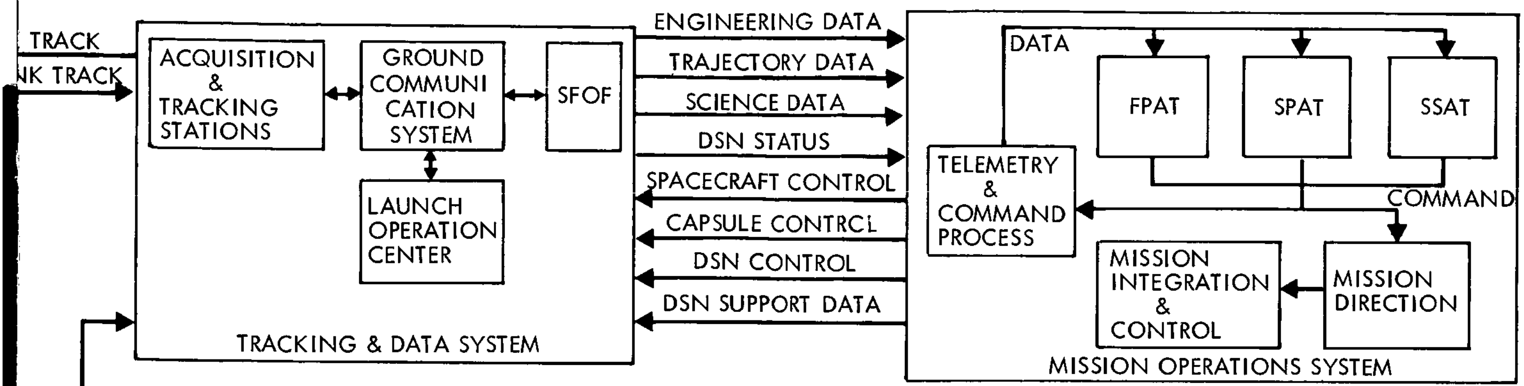


	LAUNCH VEHICLE	SPACECRAFT	CAPSULE
HARDWARE AND FACILITIES	• Multiplex for environmental sensors • VCO or transmitter for T/M pulse train	• Signal sensors, conditioners, DAE • Multiplexers, encoders, subcarrier modulators, programmers • Tape record/reproducer, buffer storage • RF receivers, transmitters, couplers, switches, antennas • Command detectors, decoders, and switching • Timing circuits	• Signal sensors and conditioners • Multiplexer and encode • Transmitter, receiver, and antennas • Tape storage device • Hardline connections (for data and comman
SOFTWARE	• Test and launch procedures • Telemetry format • Telemetry data • Tracking data • Interface definition • Functional analysis	• Transfer functions, calibrations, and error analysis • T/M format and command configuration • Operating handbooks, test, and checkout procedures • Capsule, science, and subsystem interface data	• Interface control data • Signal characteristics • Timing, multiplex, and modulation format • Calibrations and error • Test, checkout, and ope tions procedures
FUNCTIONS	• ~~Launch and Earth orbit~~ Planetary Vehicle telemetry acquisition and processing • Launch and Earth-orbit tracking	• ~~Measure physical functions and convert to T/M data~~ • Multiplex, encode all signals • Store and play back multiplex signals • Transmit data • Receive, detect, and decode commands • Receive, store, and/or transmit capsule data • Receive, translate, and transmit tracking signals	• ~~Measure and provide~~ (through hardline or RF) capsule engineering • Measure and provide (through RF) capsule science • Receive commands



	TDS	MOS
s s)	● DSIF ● AFETR ● Apollo Sites } Antennas, transmitters, and receivers; Computers, display equipment; Network processing equipment; Command and tracking equipment ● GCS (NASCOM, Goldstone/SFOF Links) ● SFOF-Network processing equipment, computers, storage, and display	● T/M subcarrier demodulators and synchronizers ● T/M data decommutator and decoder ● Command processor ● Computer interface equipment ● Predetection recording frequency converter
ra-	● System control and operating procedures ● Test and checkout procedures ● Operational capability analysis ● Schedule and priority methods ● MOS interface control and integration	● T/M and command data handling ● Flight path, spacecraft performance, and space science analysis and command computer programs ● Mission integration and control program and procedures ● Simulation and training programs and procedures ● Test and operations handbooks ● Interface control data
	● Receive and supply T/M data to the MOS ● Provide tracking information to the MOS ● Receive MOS initiated commands and transmit to spacecraft ● Provide computing, communication, storage, and display services for MOS ● Schedule and control system operation in accord with MOS requirements	● Process T/M data for use in real time, non-real time, and computer analysis ● Perform flight, spacecraft, and science analysis ● Provide control for the mission and control data for the TDS ● Process commands to the TDS and evaluate command verification

2

Figure 3.16-1: Voyager Data System

Data taken from the two Planetary Vehicles is identical except for radio frequency. This section describes nominal modes of operation--alternates can be found in the detail subsystem (Section 4.1, Volume A) and MDE discussions (Section 3.5, Volume B).

Telecommunications--Data from each Planetary Vehicle is biphase modulated on two subcarrier channels, mixed with the tracking data, and transmitted on the coherently phase-modulated S-band carrier frequency. Planetary science and capsule-entry data are biorthogonally block-encoded for transmission on the upper frequency subcarrier. Four recorders store the planetary science data output of the DAE. Playback of these recorders and from the capsule recorder is selectable. Playback data rates are 7200 bits/second for the first 80 days in orbit and 1200 bits/second thereafter.

Cruise science (real-time or recorder playback) and capsule engineering are multiplexed with spacecraft engineering for modulation on the lower-frequency subcarrier or for recording during spacecraft maneuvers. The data rates are 80 bits/second during cruise, 288 bits/second for the first 80 days in orbit, and 60 bits/second thereafter.

Upon ground reception, the data are recorded and furnished to the MDE processing equipment for subcarrier demodulation, bit synchronization, decommutation, timing generation, and serial-to-parallel conversion. The station computer processes these outputs and the tracking data for display and teletype (TTY) or high-speed data-line formats for transmission to the SFOF. The capability to edit and format display data and select priority measurements for transmission to the SFOF is essential. Following receipt at the SFOF, the data are initially processed for real-time displays and to provide alarm monitoring. The data are then transferred to disc and permanent storage for subsequent non-real-time display and analysis. Modified JPL analysis programs are used to correlate and compute data for mission and system control and evaluation.

During prelaunch, ground reception of all data is by open loop to DSIF 71 or STC hangar; engineering data is also transmitted by umbilical to the LCE. During Earth orbit, support by ETR or the Apollo global network will be required because DSIF coverage is limited. In addition, if real time data is desired during early flight phases, such as post separation or Sun acquisition, a special problem will arise (see the Telecommunications Link Analysis in Section 4.1.8). Subsequent to Sun acquisition, the DSIF stations will receive telemetry and will begin the trans-Mars operations.

Based on received tracking and telemetry data, ground commands may be issued to correct non-nominal values of flight conditions, such as trans-Mars trajectory, orbit insertion, and orbit trim, or to take malfunction correction action. The latter commands will be prescribed for given sets of failure mode diagnostics and trend predictions. When a command is initiated, SFOF computing formulates the required message for transmission. Following verification, the command is sequenced and transmitted via TTY to the appropriate DSIF, where the received message is verified and sequenced for transmission to the Planetary Vehicle. At the proper time, the command is gated to the DSIF ground equipment and again verified. The ground equipment encodes the command, modulates the carrier, and checks the command transmission. When received by the spacecraft radio system, the command is sent to the command detector whose output is the demodulated command data. These data are transmitted, through selection logic, to the command decoder. The decoder checks spacecraft address, parity, and format; decodes the data; and routes it to the DAE, C&S, or subsystem function as required by the command message. The command is also gated to the telemetry for retransmission to the ground. Proper execution is verified by observing telemetry and tracking data.

System Timing--A representative timing diagram for the data system is shown in Figure 3.16-2. The reference timing for the system is the spacecraft clock--all data are computed in spacecraft time for presentation and control. Versatility in the ground system will be necessary to merge real-time and stored data (transmitted in reverse order) from several ground acquisition sites, two spacecraft, and two capsules. This function is critical because the data require cross-correlation, and the selection of "best" data with correct time tagging is essential.

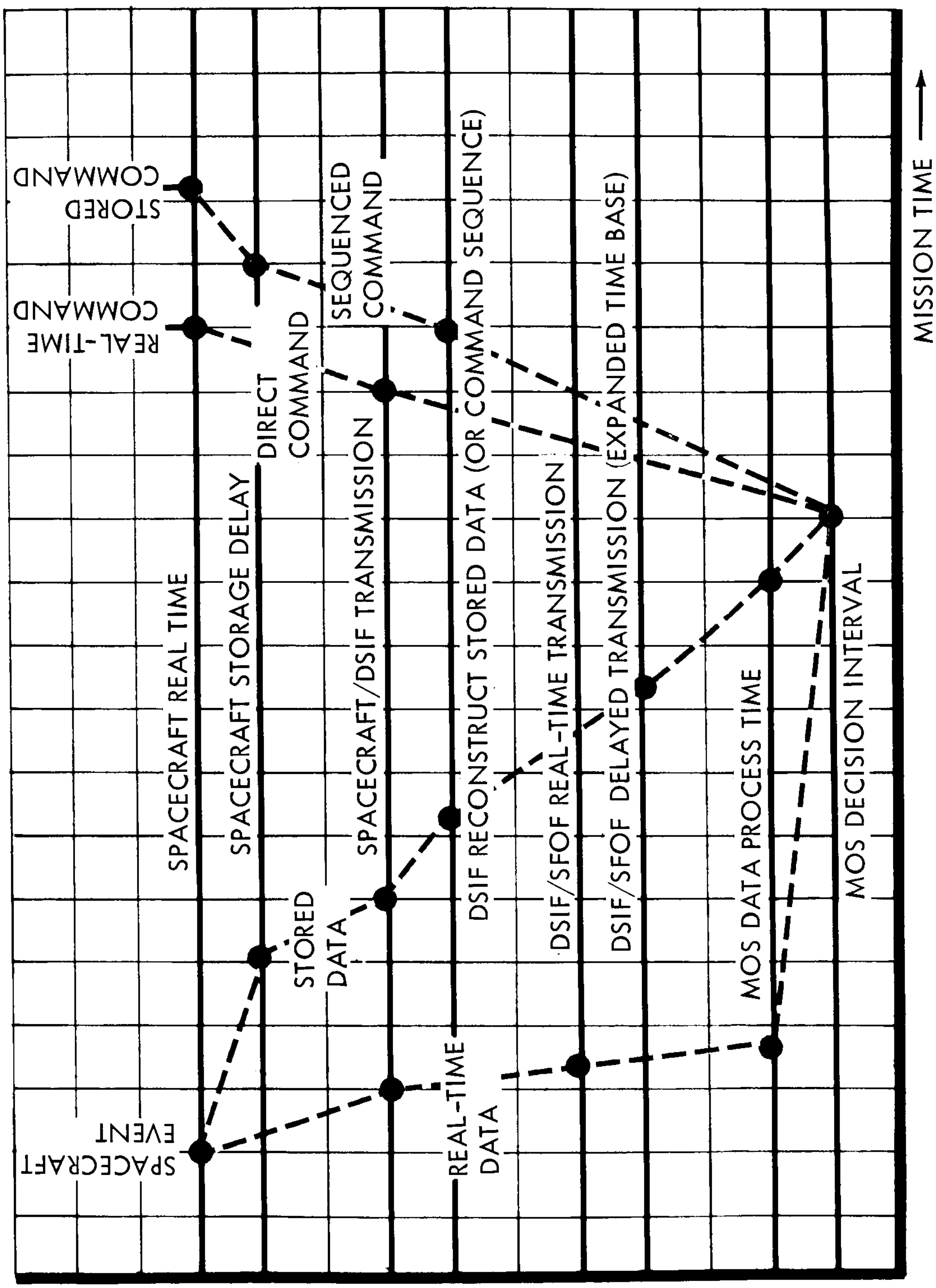


Figure 3.16-2: Data System Timing Diagram

3.17 SYSTEM RELIABILITY SUMMARY

This section introduces the governing approach to Task B reliability studies and brings together and summarizes the dispersed reliability discussions so as to give overall system clarity. Activities covered include the issuance of design requirements and the evaluations performed to determine compliance with these requirements.

The reliability approach to Task B was the same as for Task A with the following additions:

1) An increased emphasis on design reliability in all areas to prevent catastrophic failure modes, implementing preferred redundancy modes, malfunction detection, and switching techniques.
2) A detailed examination of imperative mission functions to provide a basis for maximizing the probability of mission success.
3) Examination of noncatastrophic failures to accomplish partial mission success.

Detailed backup documentation covering design data and analyses will be incorporated into the following existing documents: D2-82724-1, "Voyager Reliability"; D2-82724-2, "Voyager Failure Mode and Effects Analyses"; D2-82743-3, "Voyager Program Reliability Analyses and Prediction Standards."

Implementation of the above approach is depicted in the sequential flow diagram in Figure 3.17-1. A resume of the activity results follows:

1) Derivation, development, and issuance of reliability design requirements, objectives, and guidelines (Section 3.17.1).
2) A summary evaluation of the preferred design, including a numerical evaluation by mission phase that is further broken down to the

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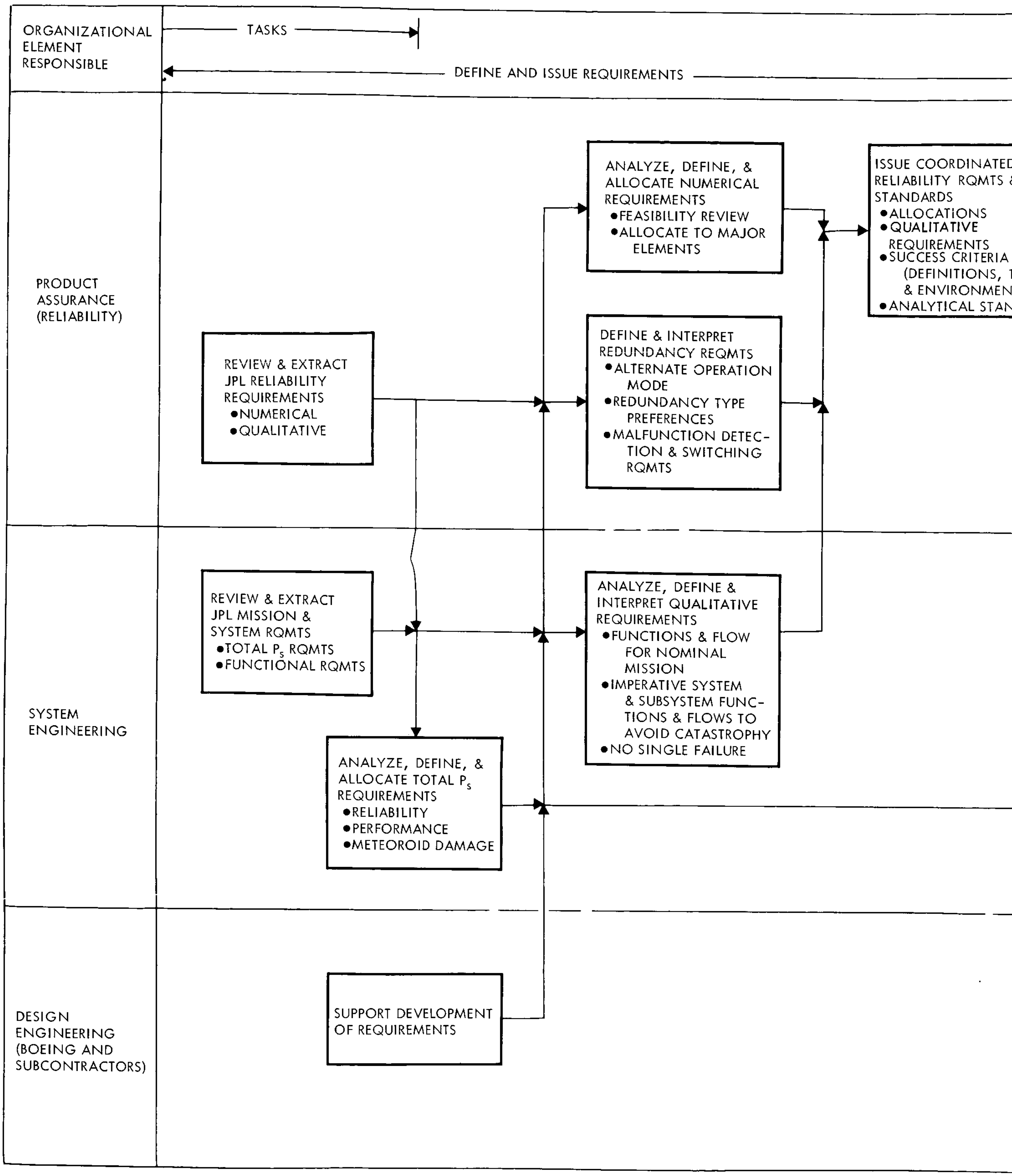

ORGANIZATIONAL ELEMENT RESPONSIBLE
TASKS
DEFINE AND ISSUE REQUIREMENTS
PRODUCT ASSURANCE (RELIABILITY)
REVIEW & EXTRACT JPL RELIABILITY REQUIREMENTS •NUMERICAL •QUALITATIVE
ANALYZE, DEFINE, & ALLOCATE NUMERICAL REQUIREMENTS •FEASIBILITY REVIEW •ALLOCATE TO MAJOR ELEMENTS
DEFINE & INTERPRET REDUNDANCY REQMTS •ALTERNATE OPERATION MODE •REDUNDANCY TYPE PREFERENCES •MALFUNCTION DETECTION & SWITCHING RQMTS
ISSUE COORDINATED RELIABILITY RQMTS & STANDARDS •ALLOCATIONS •QUALITATIVE REQUIREMENTS •SUCCESS CRITERIA (DEFINITIONS, & ENVIRONMEN •ANALYTICAL STAN
SYSTEM ENGINEERING
REVIEW & EXTRACT JPL MISSION & SYSTEM RQMTS •TOTAL P_s RQMTS •FUNCTIONAL RQMTS
ANALYZE, DEFINE & INTERPRET QUALITATIVE REQUIREMENTS •FUNCTIONS & FLOW FOR NOMINAL MISSION •IMPERATIVE SYSTEM & SUBSYSTEM FUNCTIONS & FLOWS TO AVOID CATASTROPHY •NO SINGLE FAILURE
ANALYZE, DEFINE, & ALLOCATE TOTAL P_s REQUIREMENTS •RELIABILITY •PERFORMANCE •METEOROID DAMAGE
DESIGN ENGINEERING (BOEING AND SUBCONTRACTORS)
SUPPORT DEVELOPMENT OF REQUIREMENTS

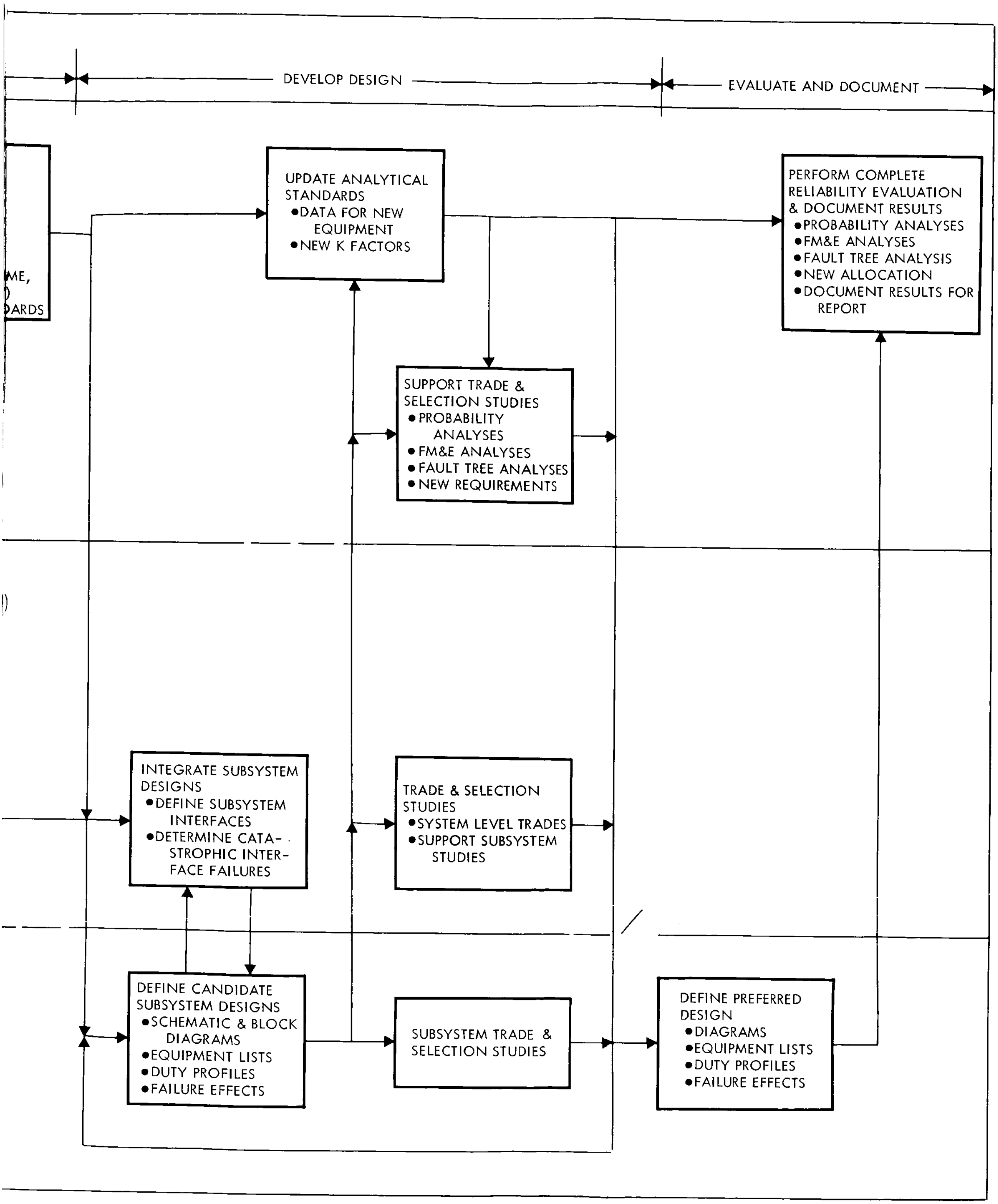


Figure 3.17-1: Task B Reliability Activity Sequence

subsystem and major component level, and a description of the methods employed to determine compliance to the "no single catastrophic failure mode" (NSCF) and partial success requirements (Section 3.17.2).

3) Detailed summaries of the reliability assessments of each subsystem of the preferred design, including numerical and qualitative evaluations, block diagrams, redundancy implementation, critical failure modes, and key reliability features (Section 3.17.3).

Although not a primary Task B requirement, brief evaluations were performed on the Saturn V launch vehicle, the Science Subsystem, the Tracking and Data System (TDS), and the Mission Operations System (MOS) equipment. Evaluations of these systems are included to show total mission relationships.

In the design area, particular effort was directed toward evolving a simple and conservative design--one whose imperative mission performance was immune to single failure. Special attention was given to developing functional independence, alternate functional paths, and a simplified method to implement alternate paths.

3.17.1 Reliability Requirements

Qualitative and quantitative reliability requirements and guidelines were developed for Task B. These requirements and guidelines were interpreted and applied to the various Voyager design areas; they served as the formal reliability direction to Task B. As standards for reliability development during Task B, two standardized missions have been adopted: a nominal mission for numerical reliability evaluations, and a minimum or "imperative" mission to assist in evaluating critical failure modes. These missions and their associated requirements are described in the following sections.

3.17.1.1 Qualitative Requirements

The objective to maximize the probability of success for the 1971 Voyager mission required the development of "hard core" mission objectives--distinct from broader mission objectives--so that imperative functions could be identified and design priorities established. Table 3.17-1 is a summary of these mission functions.

Within the above framework, the following requirements and preferences were levied on the Voyager design:

1) No single failure mode of an electronic or electrical part or component will cause the loss of an imperative function (NSCF);
2) Design simplicity will be given preference over improved performance beyond the imperative level;
3) Maximum use of conservative design (e.g., derated application, within state-of-art, proven hardware);
4) Maximum independence of functional elements;
5) Redundancy preference in the following order:
    a) Cooperative multichannel,
    b) Alternate path or functional,
    c) Block or standby;
6) Where sensing and switching is required, the following order of selection will govern:
    a) Automatic, fail-safe, on-board circuitry,
    b) Detection via telemetry--data-switch via ground command.

Table 3.17-1: MINIMUM MISSION-IMPERATIVE FUNCTIONS [1]

1) Separation of spacecraft and launch vehicle
2) Attitude reference acquisition
3) Attitude control
4) Interplanetary-trajectory correction
5) Mars-orbit injection
6) Mars-orbit correction
7) Capsule orientation and separation initiation
8) Spacecraft science data acquisition and transmission to Earth
9) Capsule science data acquisition and transmission to Earth

[1] Imperative mission functions are those functions that must be accomplished (with some possible degraded performance allowed) to preclude a catastrophic mission effect. A catastrophic mission effect is one that prevents the acquisition and transmission to Earth of scientific data by the spacecraft during Mars orbital operations. The requirement that no single failure mode of an electrical or electronic part or component will cause a catastrophic effect on the mission will, therefore, apply to each of nine imperative mission functions defined above. A further breakdown of lower-level imperative functions or events is given in Table 3.17-4.

The first selection will be used in all cases where potential failure can immediately imperil the spacecraft. The second selection may be used for potential failures that do not immediately imperil the spacecraft and only if a higher P_s results. The second method may also be used as a backup.

7) Design implementation that retains the maximum portion of system performance if a noncatastrophic failure occurs.

3.17.1.2 Numerical Requirements

Numerical reliability requirements, in the form of reliability allocations, were used during Task B to provide control of overall reliability levels, reference framework for reliability trades, and an index of complexity levels and possible problem areas.

In exercising this control, numerical evaluations were based on the nominal mission summarized in Table 3.17-2.

Table 3.17-3 lists the preliminary reliability allocations used during Task B. These allocations were derived by adjusting the revised allocation of Task A to account for: (1) the introduction of equipment to implement additional functions, (2) the scale-up of existing equipment, and (3) the introduction of additional alternate or redundant equipment.

3.17.2 Preferred System Evaluation

The system-level evaluations performed to determine compliance to established design direction are discussed below. These qualitative and quantitative evaluations were performed against the requirements and standardized missions summarized in Section 3.17.1.

Table 3.17-2: STANDARDIZED NOMINAL MISSION

MISSION PHASE	MAJOR PHASE EVENT	Δ Time (Hours)	ΣΔ Time (Hours)
Launch and Transit			
Countdown [1]	Final preparation	27	27
Launch and Injection	Boost and inject two Planetary vehicles into interplanetary transfer trajectories	1	28
Transit	Separate from launch vehicle, acquire attitude reference, and cruise including one or more trajectory-correction maneuvers	5012	5040
Planetary Vehicle Orbit Achievement	Inject Planetary Vehicle into Mars orbit, including one or more orbit-correction maneuvers	72	5112
Mars Entry [2]			
Capsule Separation	Orient the Flight Capsule and initiate capsule-spacecraft separation sequence	48	5160
Obtain Data on Mars Environment	Receive data from capsule and transmit to Earth	[3] 5	5165
Orbiter	Receive, store, and transmit data from science payload to Earth	720	5880

[1] The countdown period of the first mission phase was separated only to facilitate reliability calculations

[2] Covers spacecraft function only as indicated in event description

[3] This activity is concurrent with first part or orbiter phase.

Table 3.17-3: PRELIMINARY RELIABILITY ALLOCATION-- [1]
PHASE IA, TASK B

System Element	Allocated Reliability: System		Subsystem	
Power Subsystem			0.980	
Computing & Sequencing Subsystem			0.970 [2]	
Command Subsystem			0.970 [2]	
Guidance & Control Subsystem			0.994	
Attitude Reference				0.996
Autopilot				0.999
Reaction Control				0.999
Radio Subsystem			0.970	
Telemetry Subsystem			0.990	
Data Storage Subsystem			0.940	
Structural & Mechanical Subsystem			0.998	(0.999 ea.)
Pyrotechnics Subsystem			0.999	
Temperature Control Subsystem [4]			0.996	
Cabling Subsystem			0.999	
Spacecraft Bus (Subtotal)		0.821		
Planetary Vehicle Adapter		0.999		
Propulsion Subsystem		0.995		
Science Subsystem [3] (6 Primary Experiments)		0.800		
Reliability Contingency		0.950		
Flight Spacecraft Reliability	0.621			
Midcourse			0.997	
Orbit Injection			0.997	
Orbit Trim			0.999	
Flight Spacecraft Performance		0.993		
No Meteoroid Damage		0.990		
Flight Spacecraft P_s (Total)	0.611			
(a) P_s (at least 1 out of 2 Spacecraft)		0.849		
(b) Spacecraft OSE (Mission Critical)		0.990		
(c) Launch Vehicle P_s		0.900		
(d) MOS & TDS (Mission Critical) P_s		0.980		
(e) Launch on Time (Launch Vehicle System)		0.990		
P_s (a)x(b)x(c)x(d)x(e)	0.734			

[1] Allocations based standardized on nominal mission (see Table 3.17-2).

[2] Probability of success for commands to Spacecraft is estimated at 0.999 when considering joint capability of two subsystems. For simplicity, they are carried in series yielding a conservative P_s.

[3] Evaluation of all experiments listed in Table 3.17-8.

[4] Combination of Spacecraft Bus and Propulsion Temperature Control Subsystems.

Technical data presented includes: (1) a description of the design response and methodology employed to ensure compliance to qualitative requirements, and (2) a numerical evaluation of the preferred system shown by mission phase and further broken down to the subsystem and major-component level.

3.17.2.1 Qualitative Evaluation

NSCF Evaluation--An adaptation of the fault-tree analysis technique in conjunction with system level FM&E analyses has been used to ensure design compliance to the NSCF requirements and as a tool for controlling functional dependency. The fault-tree analysis, developed to detect and eliminate unsafe events during Minuteman mission operations, consists of relating potential failure sequences to the occurrence of an undesired event.

For the Voyager mission, undesired events are the loss of any of the imperative mission functions defined in Table 3.17-1 and illustrated in fault-tree form in Figure 3.17-2. A further breakdown is given in Table 3.17-4. This information is developed by considering each of the imperative mission functions individually and then combining the identified events into a mission sequence. Subsystem functions that are necessary for several imperative mission functions--such as "supply power" and "provide temperature control"--are not repeated, although such functions must be continued throughout the mission. For example, orbit trim is omitted because, for the preferred design, the events involved are the same as for midcourse correction. The fault-tree technique uses event logic diagrams to identify and relate basic system faults that

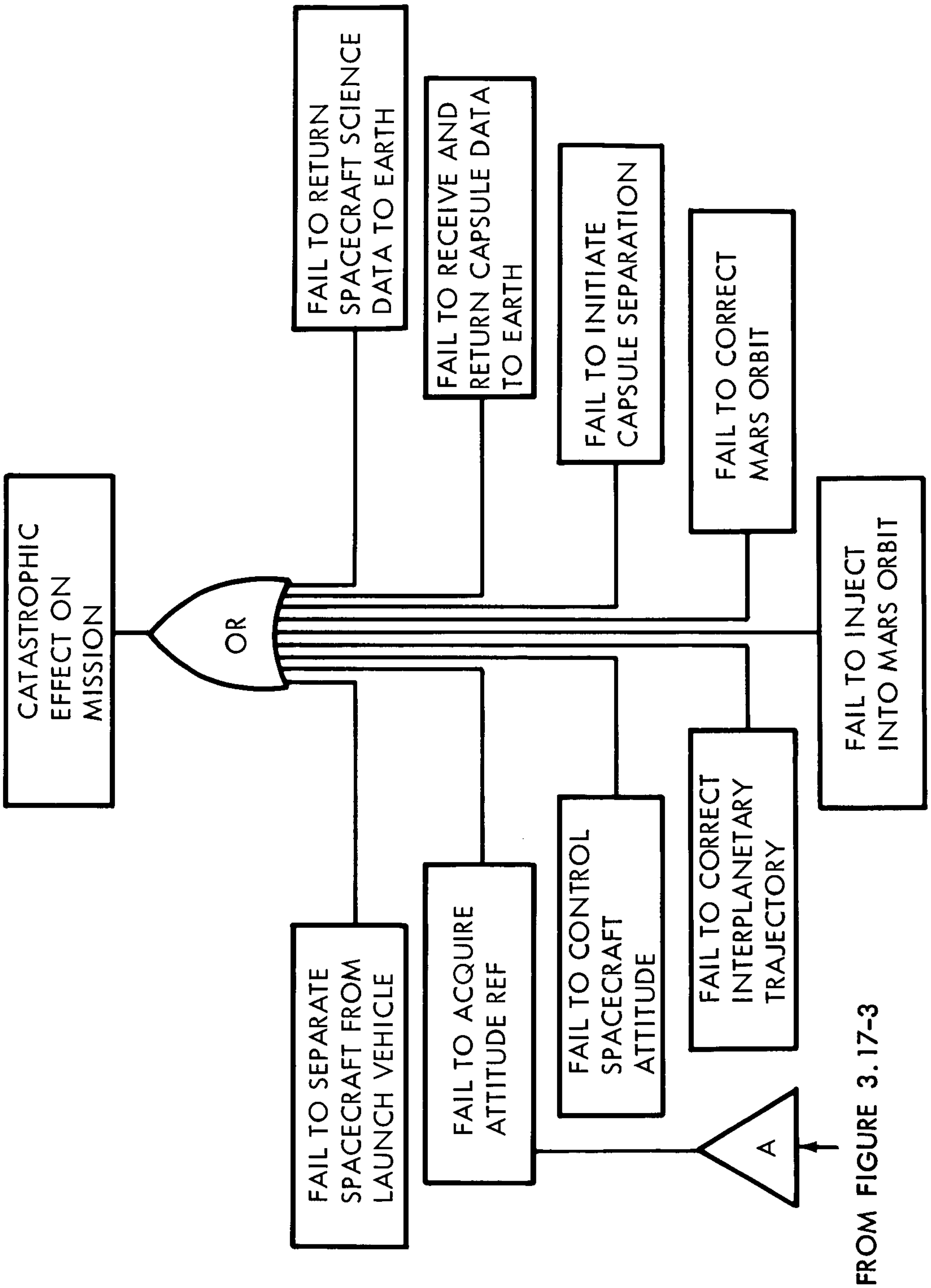


Figure 3.17.2: System Level Fault Tree for Minimum Voyager Mission

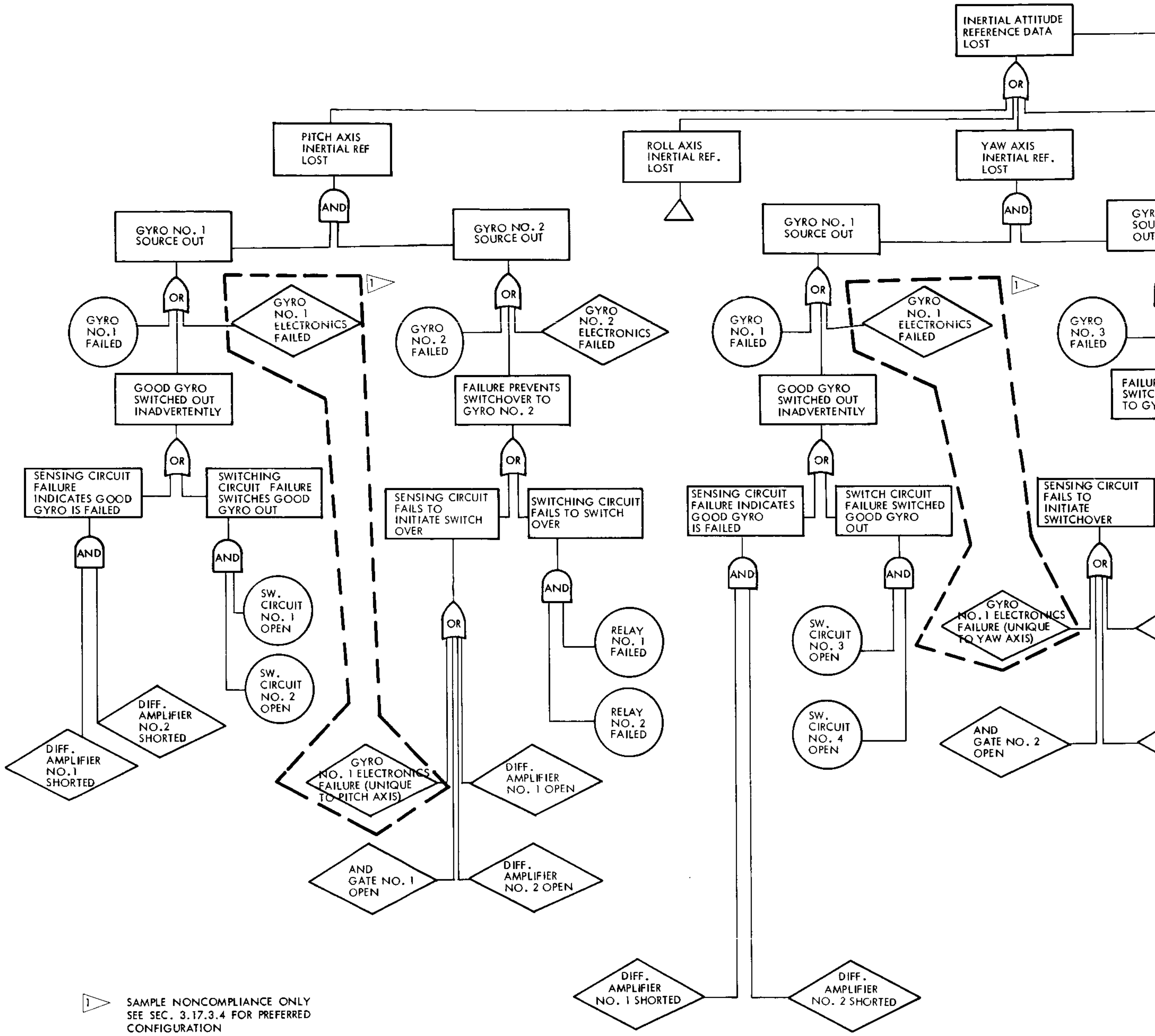
Table 3.17-4: Imperative Events

0. Prelaunch--MOS
1. Launch, injection, and shroud ejection--Launch System
2. Provide physical support for all equipment--Structures Subsystem
3. Provide and distribute electrical power--Power and Cabling Subsystems
4. Control temperature environment--Temperature Control Subsystem
5. Emit tracking signal--Radio Subsystem
6. Receive spacecraft signal and track vehicle--DSN
7. Signal to separate vehicles and enable computing and sequencing (C&S) subsystem--Launch System
8. Separate vehicle--Structures Subsystem
9. Provide antenna deployment signal to pyrotechnic--C&S Subsystem
10. Fire squibs--Pyrotechnic Subsystem
11. Deploy antennas--Structures and Mechanical Subsystem
12. Activate command subsystem--C&S Subsystem
13. Activate guidance and control subsystem--C&S Subsystem
14. Warm up and operate components--Guidance and Control Subsystem
15. Enable Sun acquisition maneuver--C&S Subsystem
16. Perform Sun acquisition maneuver--Guidance and Control Subsystem
17. Send Sun-presence signal to C&S subsystem--Guidance and Control Subsystem
18. Receive Sun-presence signal--C&S Subsystem
19. Signal guidance and control to enable roll control and turn on Canopus tracker--C&S Subsystem
20. Roll 360°--Guidance and Control Subsystem
21. Provide star-map signals to telemetry--Guidance and Control Subsystem
22. Receive and process star-map data and send to radio subsystem--Telemetry Subsystem
23. Process star-map signal and send to DSN--Radio Subsystem
24. Receive and process star map and send command to roll to Canopus--DSN
25. Receive roll to Canopus command from DSN--Radio Subsystem
26. Receive roll to Canopus command from radio subsystem--Command Subsystem
27. Roll to Canopus upon command receipt--Guidance and Control Subsystem
28. Acquire Canopus--Guidance and Control Subsystem
29. Provide Canopus-acquisition signal to C&S subsystem--Guidance and Control Subsystem
30. Transmit Canopus verification signal to telemetry subsystem --C&S Subsystem
31. Receive and process Canopus verification signal and send to radio subsystem--Telemetry Subsystem
32. Receive Canopus verification signal and transmit to DSN--Radio Subsystem
33. Receive Canopus verification--DSN
34. Maintain attitude control--Guidance and Control Subsystem
35. Send tracking signal to radio subsystem--DSN
36. Respond to DSN tracking signal--Radio Subsystem
37. Receive and process tracking signal--DSN
38. Provide MOS-updated midcourse correction definition signal --DSN
39. Receive midcourse correction message and forward to command subsystem--Radio Subsystem
40. Decode midcourse correction message and forward to C&S Subsystem--Command Subsystem
41. Store midcourse correction message--C&S Subsystem
42. Signal guidance and control to make maneuver--C&S Subsystem
43. Make attitude maneuver--Guidance and Control Subsystem
44. Enable propulsion subsystem--C&S Subsystem
45. Turn on accelerometers--C&S Subsystem
46. Signal pyrotechnic subsystem to initiate propulsion--C&S Subsystem
47. Initiate propulsion--Pyrotechnic Subsystem
48. Provide thrust--Propulsion Subsystem
49. Signal pyrotechnic subsystem to terminate thrust--Guidance and Control Subsystem
50. Terminate thrust--Pyrotechnic Subsystem
51. Signal guidance and control subsystem to maneuver and reacquire references--C&S Subsystem
52. Maneuver, reacquire references, and stabilize attitude--Guidance and Control Subsystem
53. Enable science packages and data recorders--C&S Subsystem
54. Begin operation--Data Storage Subsystem
55. Begin operation--Science Subsystem
56. Signal pyrotechnic subsystem to eject biobarrier--C&S Subsystem
57. Eject biobarrier--Pyrotechnic Subsystem
58. Transmit MOS orbit injection parameters--DSN
59. Receive orbit injection parameters--Radio Subsystem
60. Decode injection parameters--Command Subsystem
61. Store injection parameters--C&S Subsystem
62. Signal guidance and control subsystem to make preinjection attitude maneuver--C&S Subsystem
63. Make preinjection maneuver--Guidance and Control Subsystem
64. Enable Mars orbit injection engine--C&S Subsystem
65. Turn on accelerometers--C&S Subsystem
66. Signal pyrotechnic subsystem to initiate propulsion--C&S Subsystem
67. Initiate propulsion--Pyrotechnic Subsystem
68. Provide thrust--Propulsion Subsystem
69. Signal guidance and control subsystem to maneuver and reacquire references--C&S Subsystem
70. Maneuver, reqcquire references, and stabilize attitude--Guidance and Control Subsystem
71. Send MOS capsule separation signal--DSN
72. Receive capsule separation command--Radio Subsystem
73. Decode capsule separation command--Command Subsystem
74. Store capsule separation command--C&S Subsystem
75. Signal guidance and control subsystem to make precapsule separation attitude maneuver--C&S Subsystem
76. Make precapsule separation maneuver--Guidance and Control Subsystem
77. Signal capsule to separate--C&S Subsystem
78. Signal guidance and control subsystem to reorient spacecraft and stabilize attitude--C&S Subsystem
79. Receive capsule data--Radio Subsystem
80. Store capsule data--Data Storage Subsystem
81. Provide readout control signals to data storage subsystem--C&S Subsystem
82. Provide capsule data and spacecraft science data to telemetry subsystem--Data Storage Subsystem
83. Process data for transmission--Telemetry Subsystem
84. Transmit data--Radio Subsystem
85. Receive data--DSN

contribute to the end-fault condition defined by the undesired event. The fault-tree analysis is developed in terms of failures rather than successes. In a success tree, two parallel systems would feed through an "or" gate to indicate that either system can provide a success. In a fault tree, these two systems feed through an "and" gate to indicate that both systems must fail in order to cause a mission failure. Diagrams are traced downward from the undesired event on a fault tree to determine the existence of an "and" gate on each path, which indicates that a double failure must occur in order for the path to propogate toward the undesired event.

The basic technique can be expanded to any degree, allows introduction of probabilities on each event, and is amenable to computer analysis for more detailed analyses in later program phases.

Figure 3.17-3 illustrates the use of this technique to ensure compliance with the requirement that no single failure of an electrical or electronic part or component will cause the catastrophic event, "fail to acquire attitude reference." Probabilities need not be entered in the fault tree to determine compliance with qualitative requirements. However, the tree is developed to a level that ensures functional independence between two alternate function paths. For many complex systems, the initial fault tree must be developed far below the first "and" gate on each path. The tree is then solved by either a computer program or by Boolean algebra technique to determine functional dependencies between the different branches of the tree. The analysis shown in Figure 3.17-3 is not for the preferred configuration and shows some noncompliant areas for illustration purposes.



1 SAMPLE NONCOMPLIANCE ONLY SEE SEC. 3.17.3.4 FOR PREFERRED CONFIGURATION

2 SAMPLE NONCOMPLIANCE ONLY SEE SEC. 3.17.3.8 FOR PREFERRED CONFIGURATION

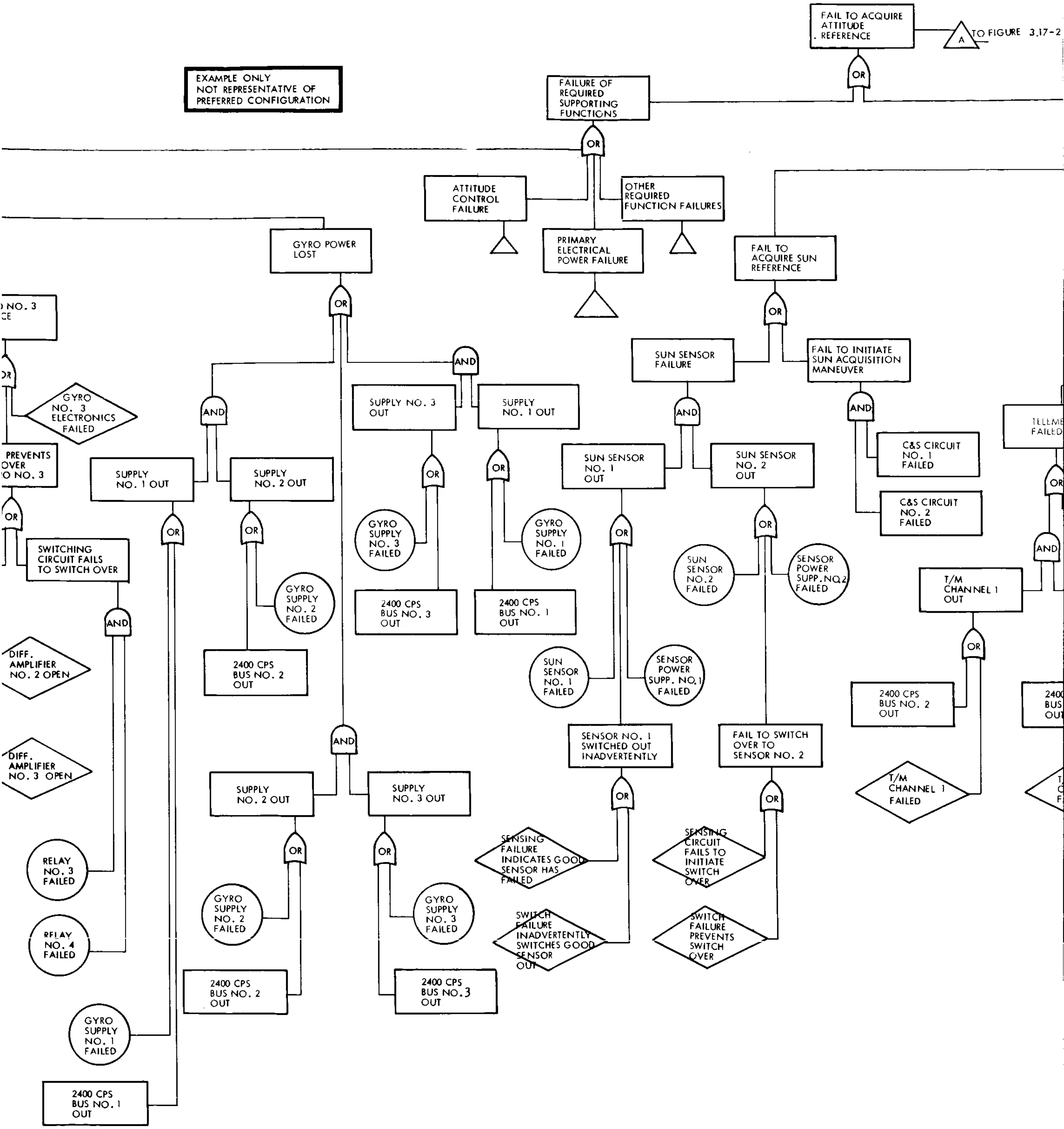
Indicates an area where a single failure will have a catastrophic effect on the mission

Indicates a conditional input to an inhibit gate

Indicates an event that results from the combination of other events through the logic gates

Indicates a basic fault event that requires no further developme[nt] includes component failures whose frequency and mode of failur[e] laboratory testing

/



AND The "and" gate describes the logical operation whereby the coexistence of all input events are required to produce the output event

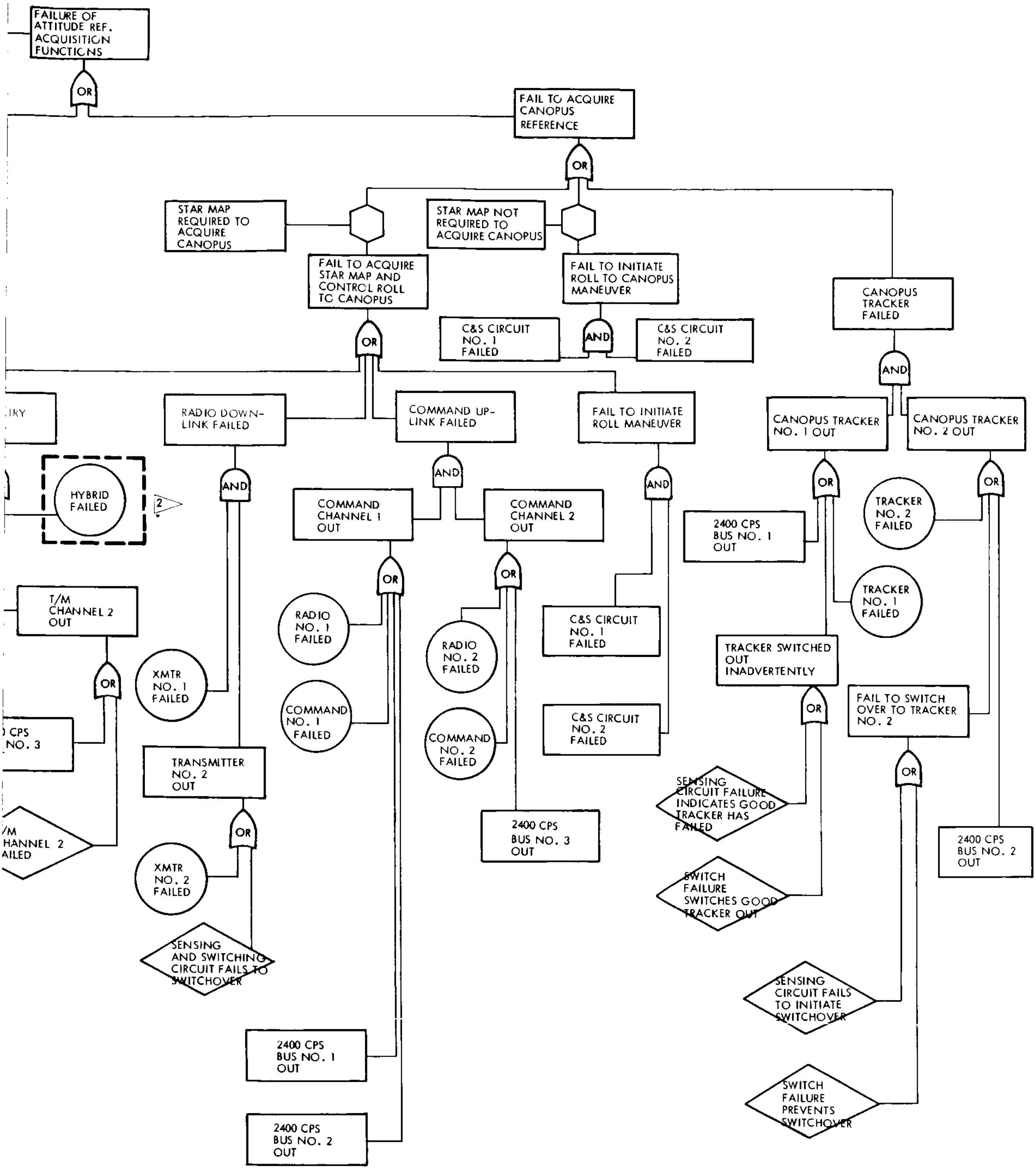
OR The "or" gate defines the situation whereby the output event will exist if any or all of the input events is present

Indicates a fault event that is considered basic... causes of the event have not been developed e... consequence or the necessary information is un...

Indicates a transfer-in of a fault tree develop...

. This category
are derived through

2



in a given fault tree; however, the
ther because the event is an insufficient
available

Indicates a transfer out to a fault tree developed on another page or location

d on another page or location

The "inhibit" gate describes a causal relationship between one fault and another or between one fault and a system condition

Figure 3.17-3: Sample Second-Level Fault Tree Logic Diagram

During Task B, informal analyses were conducted on each subsystem to the degree permitted by the design definition. Results of these analyses are discussed in more detail under the respective subsystem write-ups of Sections 3.17.3 and 4.0. The complete analysis using the fault-tree technique for each imperative mission function will be completed during subsequent program phases.

Partial Success Evaluation--One Boeing approach to the evaluation of partial success combines the results of reliability analyses, failure mode and effects analyses, and functional analyses to arrive at indices of function criticality. These are then used to indicate areas where elimination of failure modes would be most fruitful in accomplishing a maximum of mission objectives. These efforts result in reliability reallocations with concomitant design changes and weight and volume reallocations.

The criticality-analysis method is accomplished in three steps:

1) Integrate the data quality or value with the expected data quantity;
2) Determine the dependency of data objectives on selected functions;
3) Combine function-loss effects with function-loss probability.

The steps in this method are discussed in greater detail below. The sample graphs on Figure 3.17-4 clarify this approach.

Data Quantity and Quality--Each type of data sought in a mission has a predictable rate of acquisition under normal operating conditions. With these assumptions, the amount acquired of each data type can be plotted as a function of mission time.

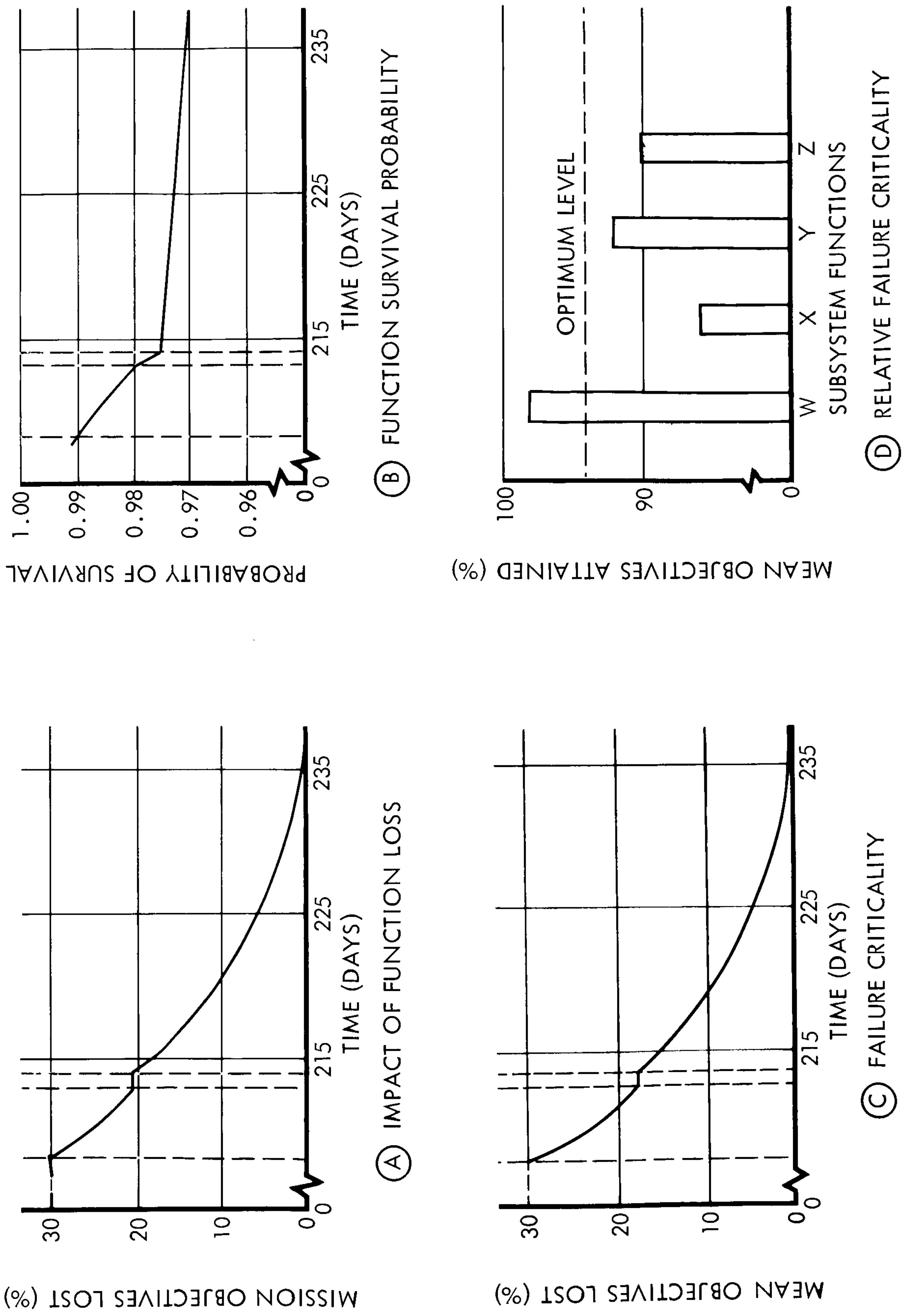


Figure 3.17-4: Criticality Analysis

Quality is a measure of the value of each type of data and varies over a period of time. For example, the TV pictures from the first spacecraft orbit have more effect than those transmitted from the 400th orbit. Such time-varying values result from customer-contractor discussion of mission objectives, just as the relative importance of each type of data is decided. Thus, data quality considers both the importance (weight) that the customer attaches to each type of data and a time variation in data value. Data-quantity and data-quality curves are then combined mathematically, point by point, for each data type to obtain a set of curves of weighted data objectives versus time.

Data Objectives and Function Definition--At any time in the mission that a selected function fails, the data objectives dependent on that function are lost from the time of failure onward. Thus, for each subsystem function, a curve can be constructed showing the proportion of weighted data objectives attained if a function failure occurs (Sketch A of Figure 3.17-4).

Effect of Function Loss and the Probability of Function Survival--The likelihood of the function failing at any point in the mission must be considered. This entails developing probability of function survival versus mission time curves (Sketch B of Figure 3.17-4). These curves are then combined point by point with the appropriate impact-of-function-loss curves to obtain a failure-criticality curve for each function (Sketch C of Figure 3.17-4). A bar chart is then prepared to compare criticality values of each function at selected points in the mission (Sketch D of Figure 3.17-4). A visual comparison of these bars indicates areas needing development attention.

Details of this method have been developed and will be incorporated in the backup documentation mentioned earlier.

Redundancy Evaluation--Evaluation of compliance to redundancy-sensing-and-switching preferences is discussed for each subsystem and major component in Sections 3.17.3 and 4.0.

3.17.2.2 Numerical Evaluation

The preferred system was evaluated using the nominal mission described in Table 3.17-2. Data standards, mathematical models, and detailed success criteria used to develop these evaluations are summarized in Section 3.17.3 and in more detail in the backup documents.

Figure 3.17-5 shows the cumulative probability of success for the preferred system as a function of mission phase. Included in the probabilities are: reliabilities of the spacecraft elements, launch vehicle, MOS and TDS, critical Flight Spacecraft performance probabilities, and probability of no meteoroid damage. The plotted path beyond the launch and injection phase is for at least one Flight Spacecraft success. Table 3.17-5 displays the end-point data as a function of system element and compares those data to the allocation given in Section 3.17.1.2.

The evaluation of the preferred system considered the partial functional redundancy existing between the computing and sequencing and command subsystems. Therefore, Table 3.17-5 shows a single evaluation that jointly covers the two subsystems. The evaluation includes all functions of both subsystems, including the effects of the partial redundancy.

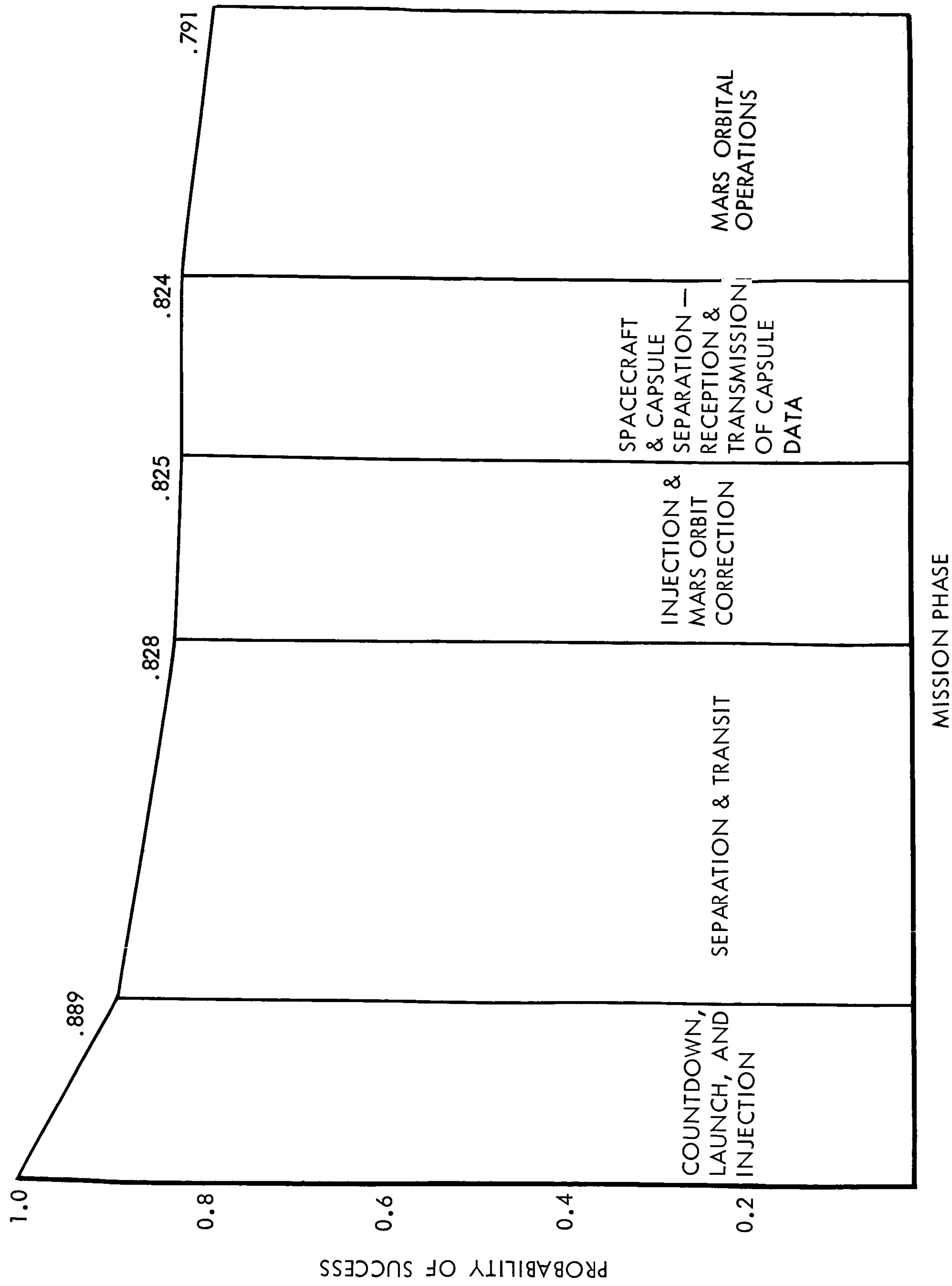


Figure 3.17-5: Cumulative Mission Success

Table 3.17-5: SUMMARY OF PREFERRED-SYSTEM RELIABILITY EVALUATION

System Element	Allocated Reliability [1]	Assessed Reliability
Power Subsystem	0.980	0.9932
Computing & Sequencing Subsystem	0.970 [2]	0.9914
Command Subsystem	0.970 [2]	
Guidance & Control Subsystem	0.994	0.9945
Radio Subsystem	0.970	0.9820
Telemetry Subsystem	0.990	0.9978
Data Storage Subsystem	0.940	0.9490
Structural & Mechanical Subsystem	0.998	0.9984
Pyrotechnics Subsystem	0.999	0.9999
Temperature Control Subsystem [4]	0.996	0.9990
Cabling Subsystem	0.999	0.9960
Spacecraft Bus	0.821	0.9045
Planetary Vehicle Adapter	0.999	0.9990
Propulsion Subsystem	0.995	0.9962
Science Subsystem (6 Primary Experiments) [3]	0.800	0.8001
Reliability Contingency	0.950	
Flight Spacecraft	0.621	0.7202
Flight Spacecraft Performance	0.993	0.9930
No Meteroid Damage	0.990	0.9900
Flight Spacecraft P_s (Total)	0.611	0.7080
P_s (at least 1 out of 2 spacecraft)	0.849	0.9147
Spacecraft OSE (Mission Critical)	0.990	0.990
Launch Vehicle P_s	0.900	0.900
Launch on Time (Launch Vehicle System)	0.990	0.990
MOS and TDS (Mission Critical) P_s	0.980	0.980
P_s (Mission Total)	0.734	0.7907

[1] Allocations based on Standardized nominal mission (see Table 3.17-2).

[2] Probability of success for commands to the Spacecraft Bus is estimated at 0.999 when considering joint capability of two subsystems. For simplicity, they are carried in series yielding conservative P_s.

[3] Evaluation of all experiments listed in Table 3.17-8.

[4] Combination of Spacecraft Bus and Propulsion Temperature Control Subsystems.

A comparison of the allocated and assessed values on Table 3.17-5 shows that the preferred-system reliability and P_s exceed the objectives set forth at the beginning of Task B. These assessments will serve as a basis for a revised allocation to be used during follow-on program phases.

3.17.3 Subsystem Evaluation Summary

Evaluation results are organized by individual subsystem, with each section containing material relating to:

1) Reliability expressed as a function of contained components and mission phase;
2) Failure mode and effects analysis summary and critical life characteristics;
3) Key reliability features, including a resume of redundancy employed;
4) Summary evaluations of alternate configurations.

The major elements of guidance and control, attitude reference, autopilot, and reaction control have been treated as individual subsystems for this section of the system reliability summary. In addition, discussions of the planet sensor, guidance scan platform, limb and terminator detectors, and pointing controllers (high-gain antenna) are incorporated with the attitude reference subsystem write-up.

Substantiating data are contained in Section 4.0 and the series of referenced backup documents.

3.17.3.1 Power Subsystem

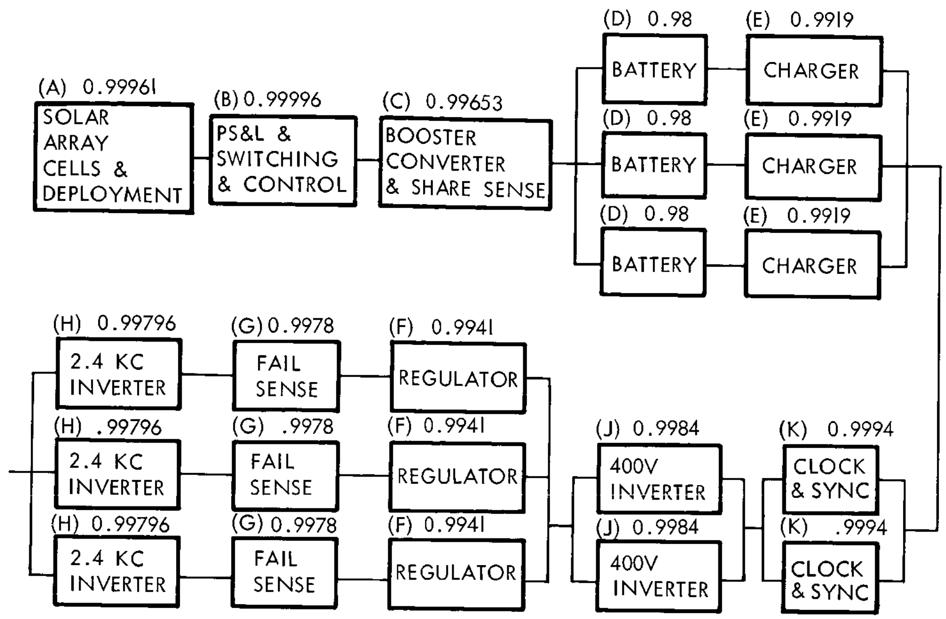
The reliability allocation for the power subsystem was 0.980. The preferred configuration, which is the result of a trade study involving seven configurations, has an assessed reliability of 0.9932. Figure 3.17-6 shows component reliabilities and redundancy schemes in a reliability block diagram, along with the mission phase hazard plot and the reliability mathematical model.

A fault-tree analysis and failure-mode and effects analysis (documented in Section 4.0) were conducted to determine the effects of failure and to establish a design that eliminates the effect of any single electronic part failure on mission success. The preferred subsystem incorporates several techniques of redundancy, including overdesign of solar arrays, active parallel components, and voting logic. Selective use was made of devices for failure sensing and switching, current limiting, and circuit overload protection with reclose capability. Power is available to recipient subsystems through a triredundant bus configuration whereby each subsystem is connected to two independent buses. The Flight Capsule derives power from two redundant, raw d.c. circuits.

The power subsystem components, assembly, and installation were analyzed for life-limiting characteristics. Battery charge-discharge cycles are well within state-of-the-art capabilities for battery life. All batteries are of the hermetically sealed type to eliminate reduction in battery life because of electrolyte loss. Overdesign of solar array panels gives extended panel life where meteoroid damage is considered the major cause of failure.

3.17.3.2 Computing and Sequencing (C&S) Subsystem

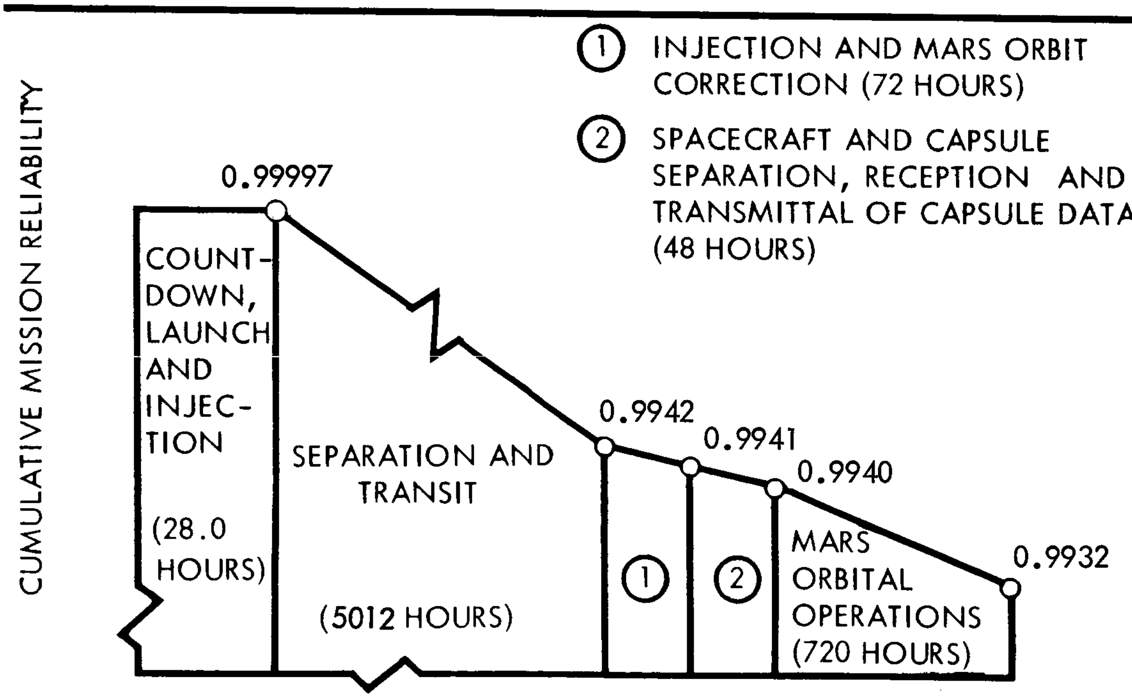
The allocated reliability for the C&S subsystem was 0.970. A preferred configuration was evolved with an assessed reliability of 0.9860.



POWER SUBSYSTEM
RELIABILITY BLOCK DIAGRAM

$$R_S = R_A R_B R_C \left[(R_D R_E)^3 + 3(R_D R_E)^2 Q_{DE}\right]\left[(R_F R_G R_H)^3 + 3(R_F R_G R_H)^2 Q_{FGH}\right]\left[1-(1-R_J)^2\right]\left[1-(1-R_K)^2\right]$$

MATHEMATICAL MODEL



MISSION PHASE HAZARD CHART

Figure 3.17-6: Preferred Power Subsystem Reliability Summary

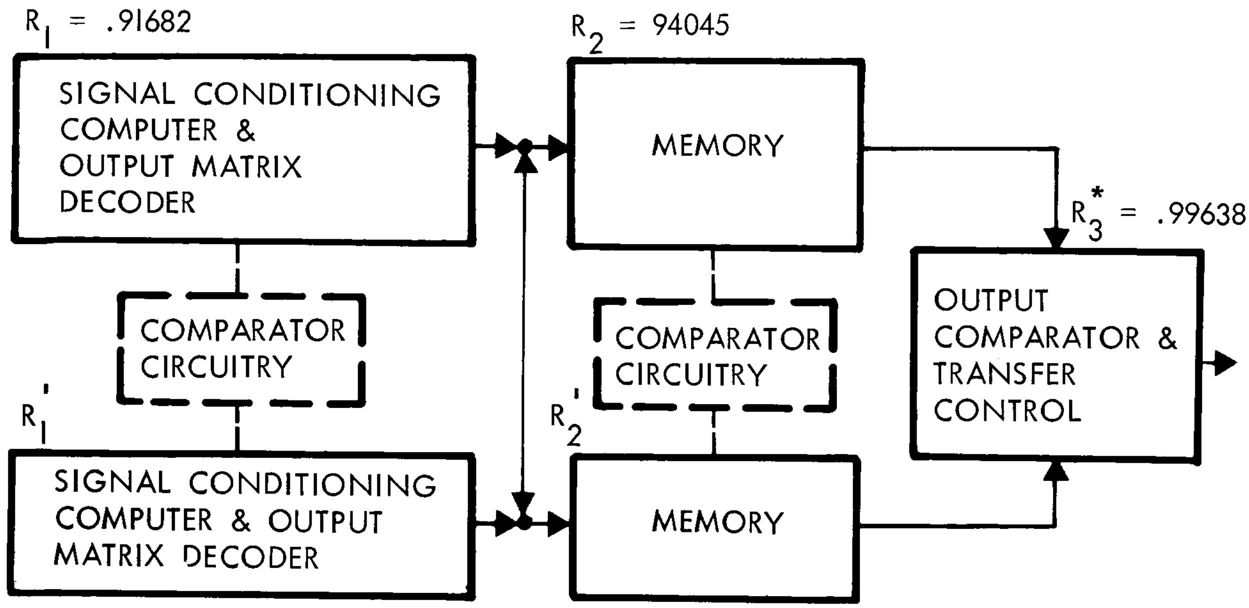
The C&S subsystem reliability block diagram is given in Figure 3.17-7 along with the mission hazard curve.

The C&S subsystem will provide all command operational signals automatically from launch to end of mission without using ground commands other than those required for interplanetary-correction or orbit-trim maneuvers. The command subsystem, which provides updating of the C&S prior to interplanetary maneuvers, also acts as backup to all recipient subsystem inputs and outputs during nonsequential modes of operation.

The preferred C&S subsystem was selected for its capability to supply subsystem operational signals through multiple paths. Redundant computers, operating synchronously in parallel with internal cross checking, ensure against mission loss from a single failure and meet all special operating requirements.

Solid-state electronic comparators are used to sense and initiate data transfer if a failure occurs. Reliability of the comparator circuitry (dotted lines in the block diagram) was considered as an integral part of the computers and memory storage units. There are no external sensing-and-switching devices required in the C&S subsystem.

A total of five configurations were analyzed, each having varying degrees of redundancy. The use of two independent, parallel, synchronous processors with an estimated total reliability of 0.99 was rejected because erroneous signals could be inadvertently generated and transmitted to a subsystem. A triplicated subsystem with parallel majority voting gates having an estimated reliability of 0.972 was also rejected.



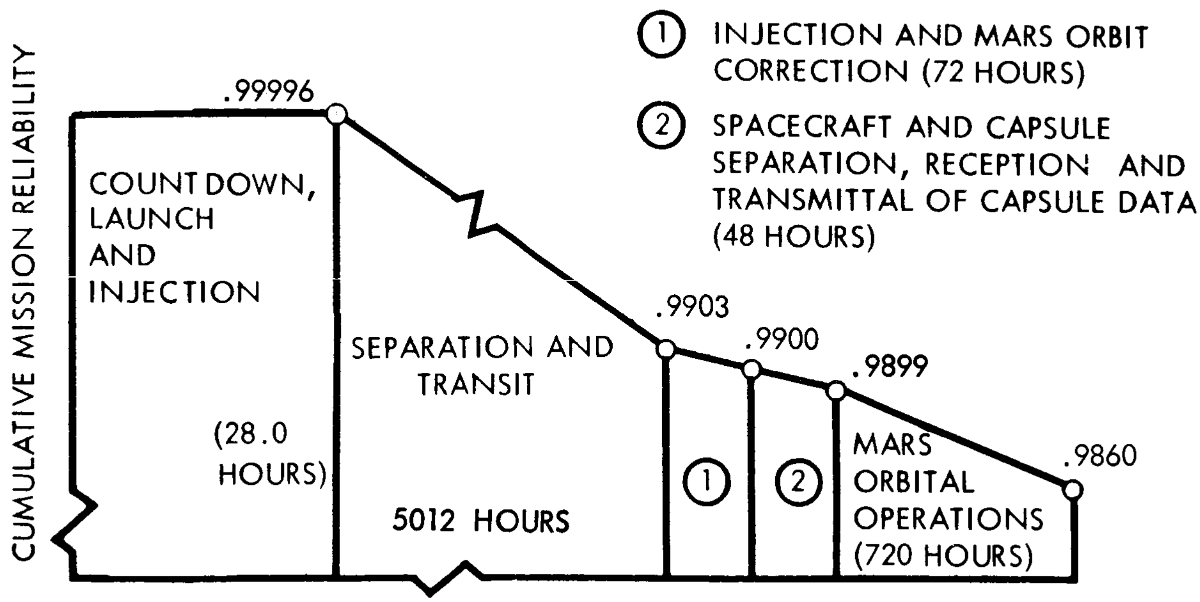
$$R_T = \left[R_1^2 R_2^2 + 2R_1^2 R_2 Q_2 + 2R_1 Q_1 R_2^2 + 4R_1 R_2 Q_1 Q_2\right] (R_3)$$

where $R_1 = R_1'$; $R_2 = R_2'$; $Q_1 = Q_1'$; $Q_2 = Q_2'$

NOTE: RELIABILITY VALUES ARE BASED ON A NOMINAL MISSION TIME PERIOD.

*INTERNAL MULTIPLE PATH REDUNDANCY

COMPUTING AND SEQUENCING SUBSYSTEM RELIABILITY BLOCK DIAGRAM



MISSION PHASE HAZARD CHART

Figure 3.17-7: Preferred Computing And Sequencing Subsystem Reliability Summary

This mechanization called for two of the three computer and sequencers to be operable at all times. A third configuration using standby redundancy was rejected because of mission specification requirements. Another alternate configuration using quad redundancy logic was discarded because of excessive weight, electrical power, and overall costs.

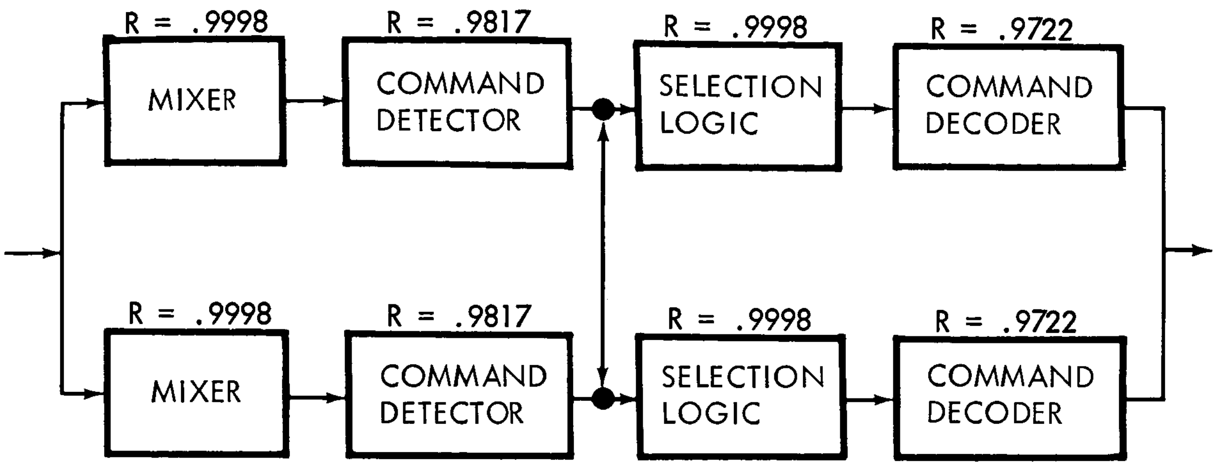
A review of the preferred subsystem equipments indicates that there are no life-limiting characteristics that would cause malfunctions or have a significant effect on mission success.

3.17.3.3 Command Subsystem

The reliability allocation for the command subsystem was 0.97, and, by using specific redundancy techniques, an assessed value of 0.9847 for all command capability was achieved. The preferred-subsystem simplified reliability block diagram and mission-hazard curve are given in Figure 3.17-8.

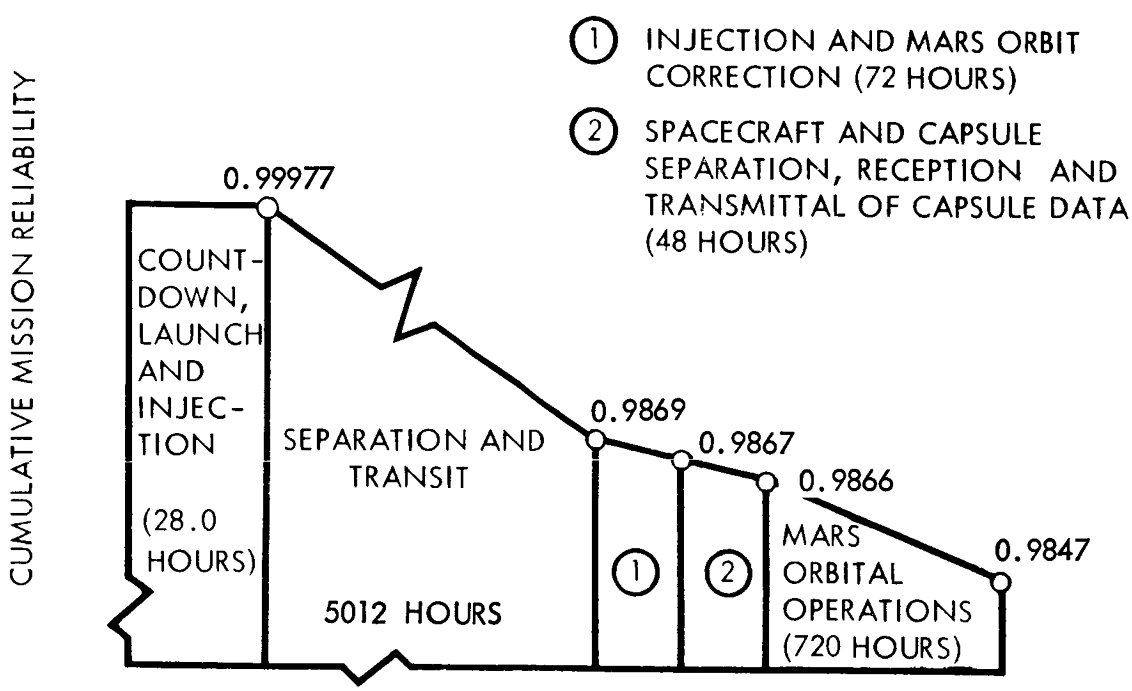
The preferred configuration (see design section for details) comprises two complete, parallel command detectors and decoders with selection logic that permits either detector to operate with either decoder (that is, the subsystem will successfully operate if both a detector and a decoder fail).

To reduce interface complexity and cable weight, the redundant decoder outputs, through a combiner circuit, become a single-thread interface with the other spacecraft subsystems. This is permissible because the command decoding functions of the command subsystem is primarily a backup for commands normally executed by the C&S subsystem.



COMMAND SUBSYSTEM
RELIABILITY BLOCK DIAGRAM

NOTE: RELIABILITY VALUES ARE BASED ON A NOMINAL MISSION TIME PERIOD



MISSION PHASE HAZARD CHART

Figure 3.17-8: Preferred Command Subsystem Reliability Summary

Another feature of the subsystem design is that the issuance of erroneous commands from a single failure is prohibited by a combination of dual operation codes and parity checking. In terms of its primary (update C&S) and secondary functions, a detailed assessment of the preferred design yielded reliabilities of 0.9999 and 0.9872, respectively. The probability to update the data automation equipment is also 0.9999.

Reliability of command functions has been augmented by:

1) The capability of the C&S subsystem or the command subsystem to actuate a given command independently,
2) The capability to gain access to spacecraft subsystems through either of the redundant radio-subsystem receivers and redundant low-gain antennas ;
3) The capability of either decoder to accept command words from either detector.

Three candidate configurations were evaluated. The single-thread design resulted in a reliability level of 0.9436; an independent alternate path concept (i.e., no crossover between decoders was assessed at 0.9842) and the preferred configuration shown in the reliability block diagram in Figure 3.17-8 was assessed at 0.9847.

The preferred-subsystem equipment circuit mechanizations were reviewed. They show that there are no life-limiting parts or components that would significantly affect the overall mission.

3.17.3.4 Attitude-Reference Subsystem

The initial reliability allocation for this subsystem was 0.996 versus a predicted reliability of 0.9965. Figure 3.17-9 shows the reliability hazard over the course of the mission. Specific subsystem components and the method of redundancy mechanization are discussed below. The system has been mechanized throughout to satisfy the requirement that no single failure mode will have a catastrophic mission effect.

The three dual-axis gyros are mounted orthogonally so that all three axes of attitude data can be obtained from any two gyros. All gyros are operated continuously, and any failed gyro can be removed simply by shutting off the power.

Catastrophic failures are detected by monitoring the speed control and pickoff-angle signals. Excessive drift is detected by comparing the value of signals for the same axis from different gyros, since two axes are uniquely defined by each gyro.

Any gyro or torque-loop failure will show up in both axes of the gyro because a drift in one axis will show up as an error in the other. The integrator is the single point at which a failure will not show in both axes. These integrators have been designed so that any failure will be of the "open" or "no output" variety, so that this failure will not be catastrophic but will only show as a reduction in gain.

The accelerometer is only included for improved accuracy and is backed up by a redundant timing signal in the C&S, which provides minimum and maximum thrust termination times for the midcourse-correction and orbit-insertion maneuvers. The C&S timing signal is the controlling function in the case of accelerometer malfunction.

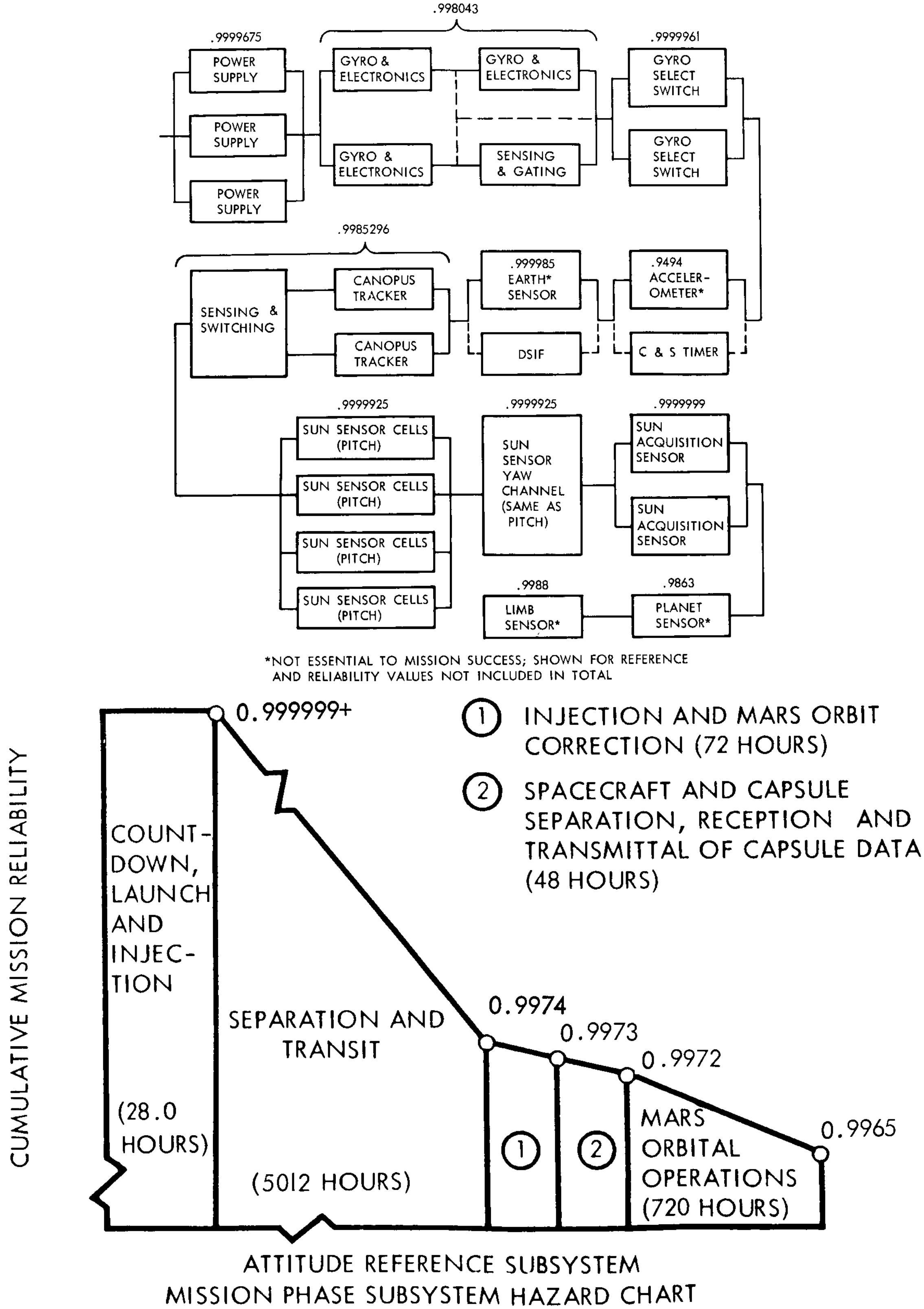


Figure 3.17-9: Preferred Attitude Reference Subsystem Reliability Summary

components and mechanization approaches are presented in Section 4.1. There are no life-limiting characteristics in the equipment that would cause premature mission termination.

3.17.3.5 Autopilot Subsystem

The allocated reliability for the autopilot subsystem was 0.999. As mechanized, the system consists of two identical systems with polarity-splitting of the electronics to isolate failures to one-half a single system. This approach uses cooperative multichannel redundancy so that a single failure will result in, at most, a loss of only half the system. Even if this loss occurs, the other half can maintain control and will have half the response rate. For simplicity, the reliability block diagram and reliability predictions for this system (Figure 3.17-10) includes only the autopilot functions. A complete system reliability block diagram would show the actual mechanization of independent channel isolation through the N_2 jets, jet vanes, and secondary injection components of the reaction control subsystem.

The Sun sensor has four redundant cadmium sulfide cells that interface independently with the four independent autopilot channels. Failure of any one of the cadmium sulfide cells will have the same system effect as a failure in one of the redundant autopilot channels. (See below)

The Canopus sensors use two identical Barnes (NASA/JPL design) units similiar to those used on Mariner IV. On-board failure detection and switching has been employed, with the constraint that a detector or switch failure cannot cause a catastrophic mission effect. Monitored failure indications are:

1) A loss of acquisition signal,

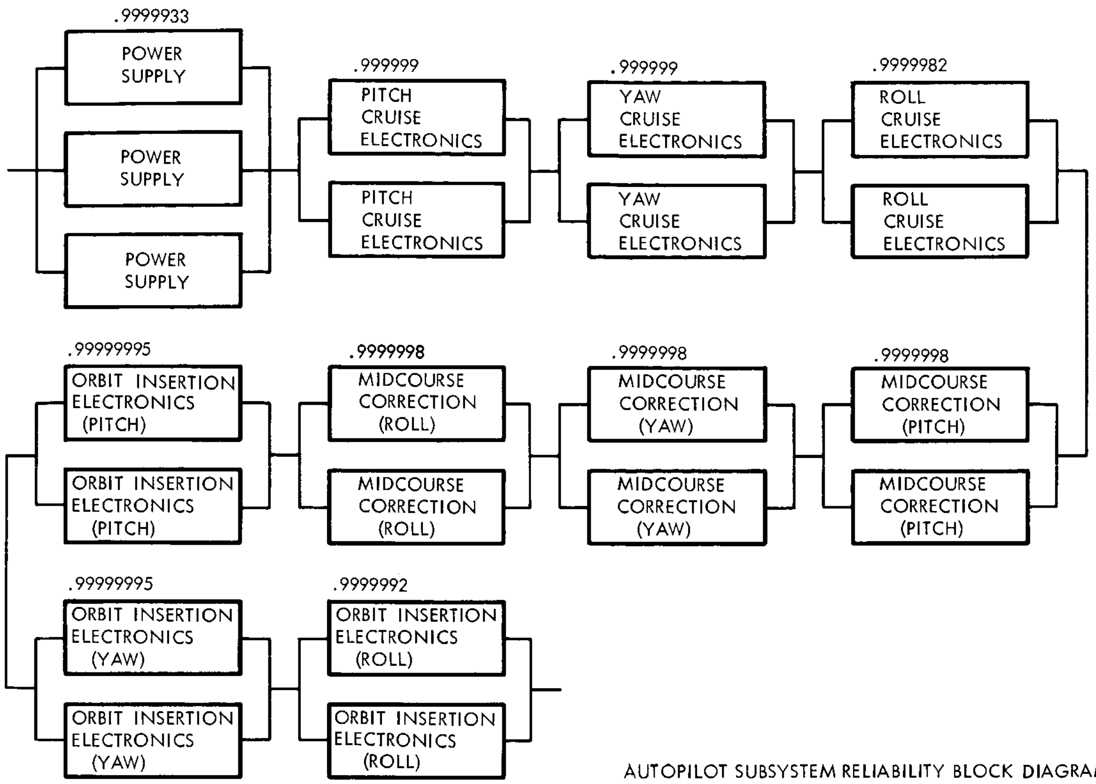
2) Occurrence of a large error signal without loss of acquisition.

Both conditions will cause switching to the redundant tracker, except when inhibited during occultation or gyro-hold modes of operation. Switching from either sensor to the other can also be performed by ground command.

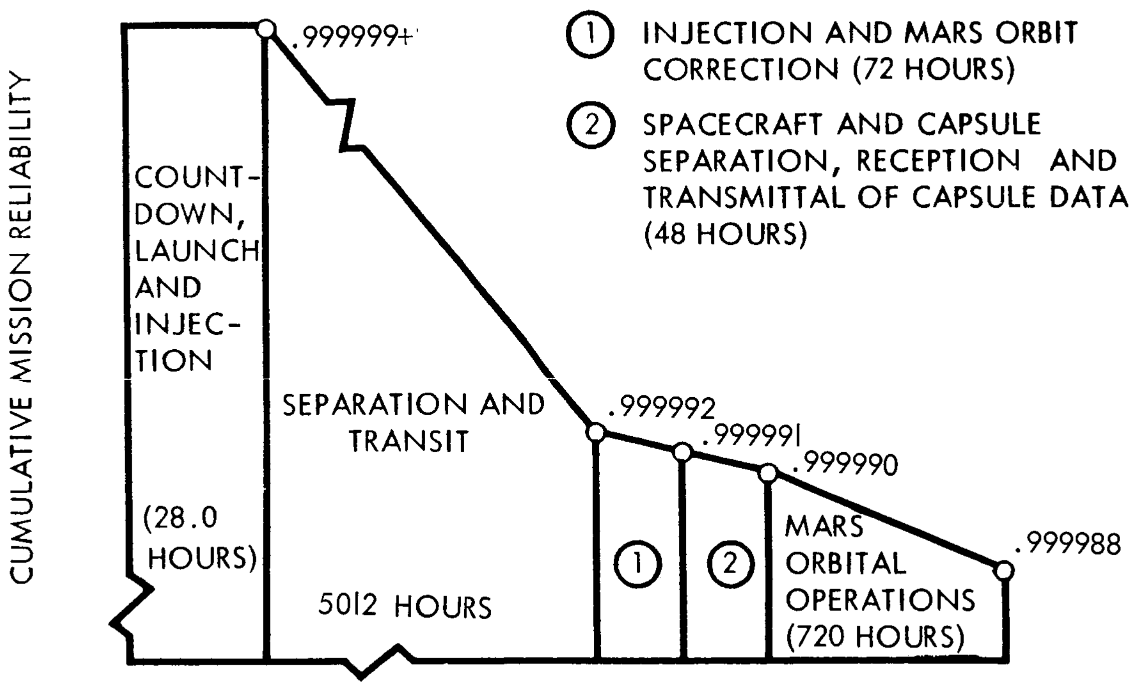
The Earth sensor has been incorporated as an additional method of verifying Canopus acquisition. It is not considered essential to mission success; therefore, redundancy was not considered necessary.

The guidance scan platform planet tracker and limb and terminator sensor unit make up the Mars sensors. Both are desirable for optimum mission performance, but neither is essential for mission success. This unit has a reliability of 0.9851.

Figure 3.17-9 shows the reliability block diagram depicting the above relationships. Specific trade-offs involved in selecting the above



AUTOPILOT SUBSYSTEM RELIABILITY BLOCK DIAGRAM



MISSION PHASE HAZARD CHART

Figure 3.17-10: Preferred Autopilot Subsystem Reliability Summary

3.17.3.6 Reaction Control

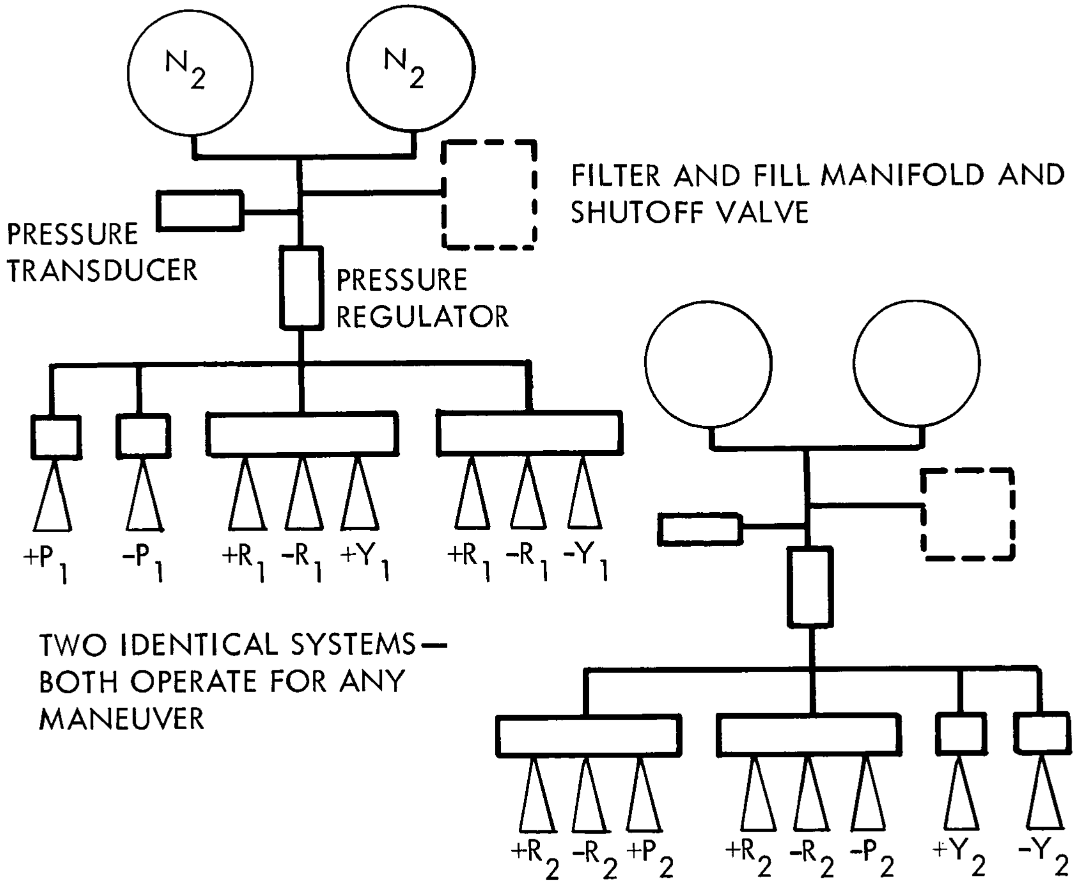
The initial reliability allocation of 0.999 was based on a projection of Phase IA allocations and assessments. The selected subsystem evaluation, summarized in Figure 3.17-11, shows a subsystem reliability of 0.998+. This reliability is broken down by mission phase in the mission-hazard curve.

Experience with similar systems on the Mariner, Ranger, OGO, OSE, OAO, and Syncom vehicles has furnished design direction for this subsystem and an indication of expected reliability.

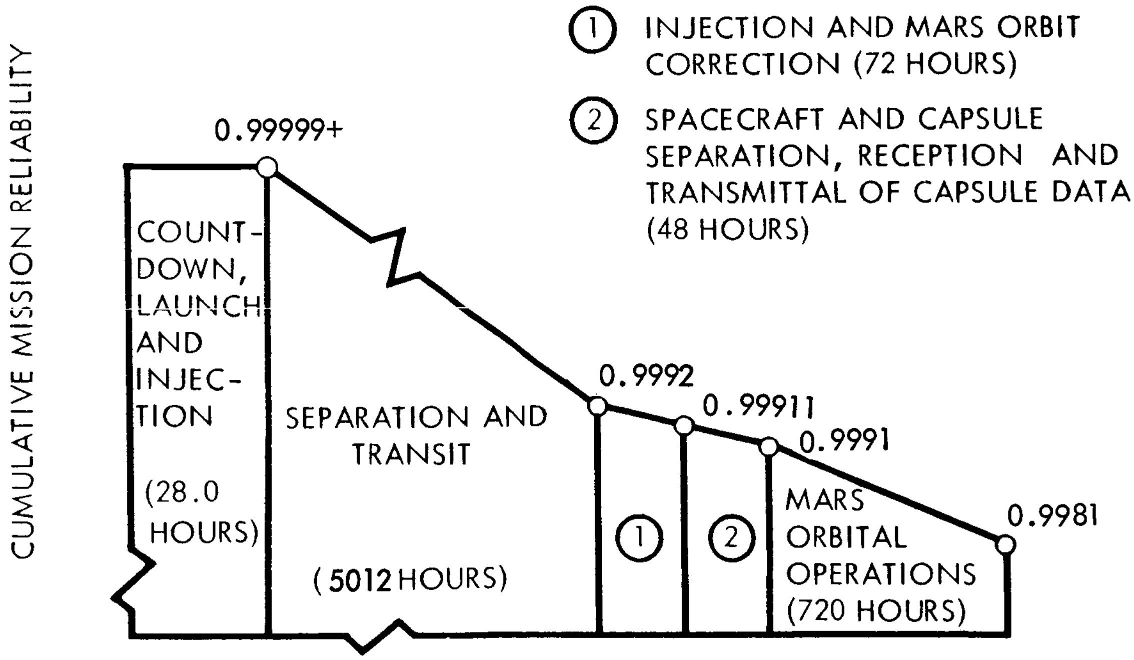
The selected subsystem uses a cooperative multichannel approach to gain redundancy. The system comprises two separate thruster systems, each containing 1.5 times the gas requirement for the total mission and each functioning during any maneuver (Figure 3.17-11). This subsystem configuration provides alternate paths of functional operation for all but one failure mode.

One trade study involved the capability to switch a lower-level pressure regulator into each N_2 feed system. This appreciably reduced the average number of cycles per thruster, which resulted in a reliability assessment of 0.999 for the subsystem. However, such a configuration change involves the addition of pyrotechnics.

A highly desirable feature of the preferred subsystem is that it does not require malfunction detection and switching equipment.



PREFERRED SUBSYSTEM FUNCTIONAL BLOCK DIAGRAM



MISSION PHASE HAZARD CHART

Figure 3.17-11: Preferred Reaction Control Subsystem Reliability Summary

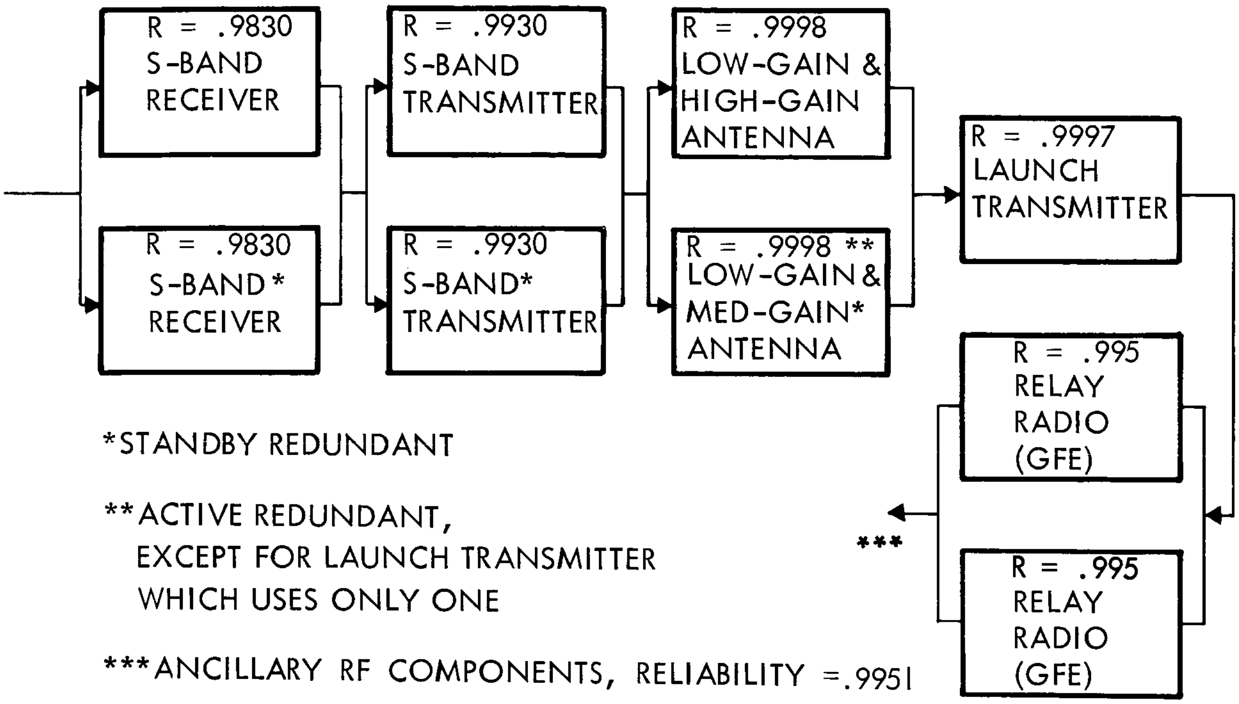
Life-limiting characteristics include: (1) the total supply of N_2 and the leak rate of the system, (2) the number of cycles of the thruster valves and pressure regulators, and (3) contamination introduced into the plumbing during fabrication. These items affect total system life; however, they are not expected to significantly degrade the probability of success for the Voyager mission. These characteristics will become more significant as the projected missions become longer.

An extremely rapid leak, including tank rupture, is the only nonredundant failure mode for the subsystem. To minimize this type of failure, a relatively high safety factor is incorporated in the design, and rigorous quality control and proof testing are incorporated in the production of the pressure vessels.

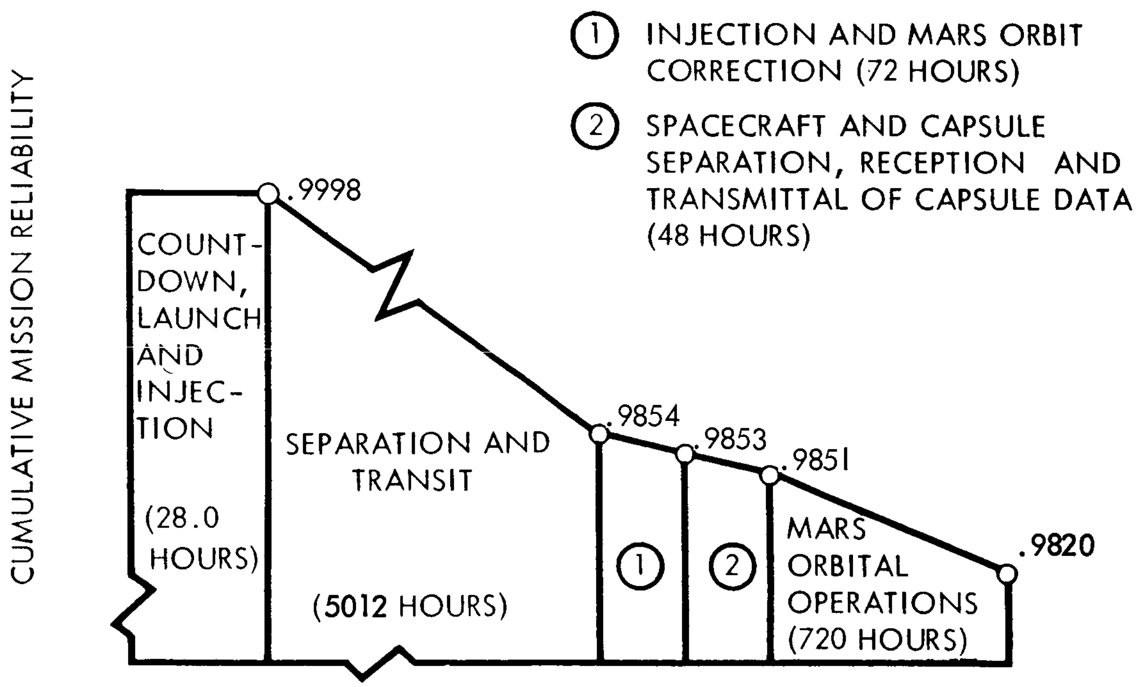
3.17.3.7 Radio Subsystem

The radio subsystem was allocated a reliability of 0.970. The reliaability assessment of 0.9820 for the preferred-design configuration is further displayed on Figure 3.17-12. Although the relay radio (GFE) was not evaluated in detail, its reliability is expected to be approximately as shown in the summary figure.

Extensive functional redundancy ensures reliability of the radio subsystem. Switching capability provides functional cross-connect flexibility to ensure operating mode reliability. Provision for backup modes allows additional success expectancy in acceptable, although slightly degraded, modes. For example, two antennas transmit telemetry



RADIO SUBSYSTEM RELIABILITY BLOCK DIAGRAM



MISSION PHASE HAZARD CHART

Figure 3.17-12: Preferred Radio Subsystem Reliability Summary and ranging data. Redundancy control is provided by sensing and automatic switching plus programmed sequence instructions from the computer and sequencing subsystem for mode initiation at the appropriate mission point. The redundancy control will also accept ground-station commands to enhance mode initiation, as required.

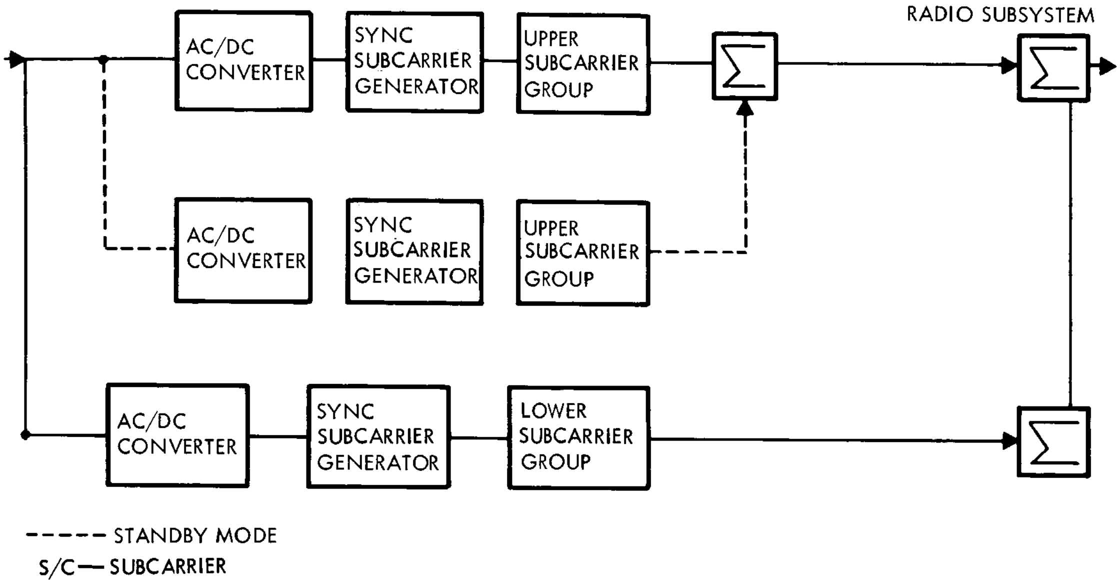
Several trade studies were performed to determine the appropriate balance between performance and reliability. A typical study evaluated alternate S-band receivers, so that the effect of either active or standby redundancy in the receiver circuitry could be estimated.

Based on the proposed reliability program disciplines and the proposed design, no parts or materials used in the radio subsystem will exhibit life-limiting failure mechanisms during the mission. The traveling-wave tube amplifier normally is sensitive to failure because of cathode depletion. Although no life data are available on TWT's in the 50- to 100-watt category, life data on 10- to 13-watt devices indicate lifetime of about 30,000 hours. Because there is a close generic relationship between the 10- to 13-watt devices and the higher-power tubes, it is expected that the Voyager tube will exhibit suitable lifetime characteristics.

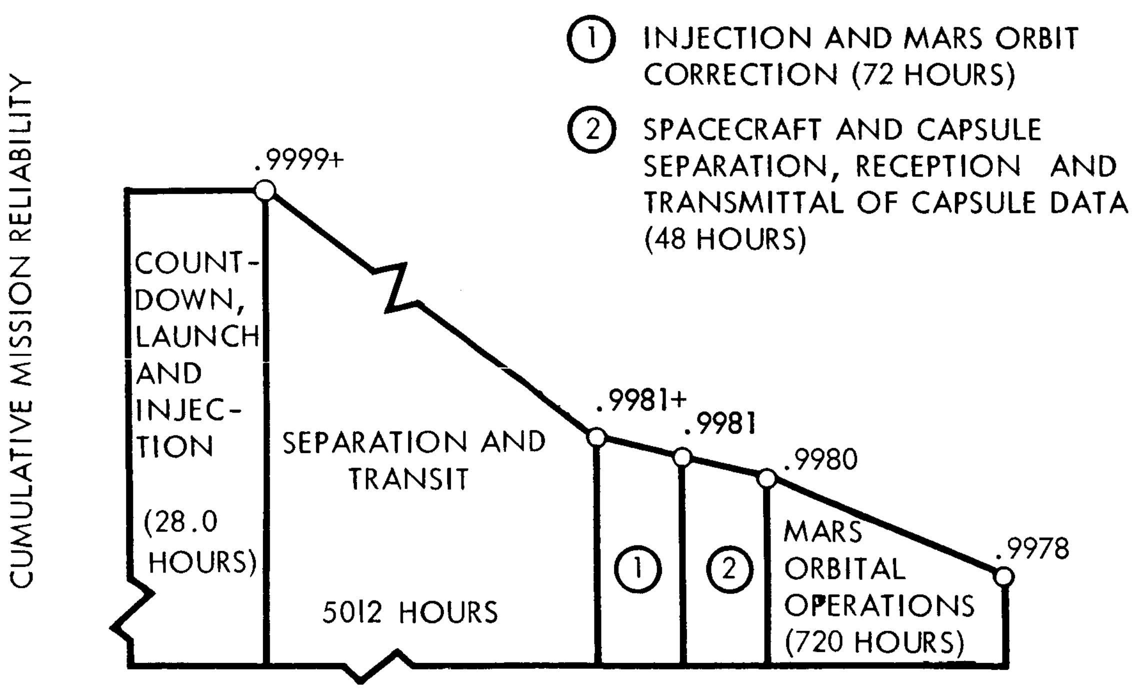
3.17.3.8 Telemetry Subsystem

The telemetry subsystem was allotted a reliability of 0.990. Based on the reliability block diagram in Figure 3.17-13, the preferred design configuration is assessed to have a reliability of 0.9978.

The functional alternate path concept ensures reliability for these



TELEMETRY SUBSYSTEM
RELIABILITY BLOCK DIAGRAM



MISSION PHASE HAZARD CHART

Figure 3.17-13: Preferred Telemetry Subsystem Reliability Summary functions (i.e., the block-encoded upper-subcarrier planetary science data and the lower-subcarrier cruise science, flare, capsule, engineering, and maneuver data are functionally independent). The upper channel is also fully redundant.

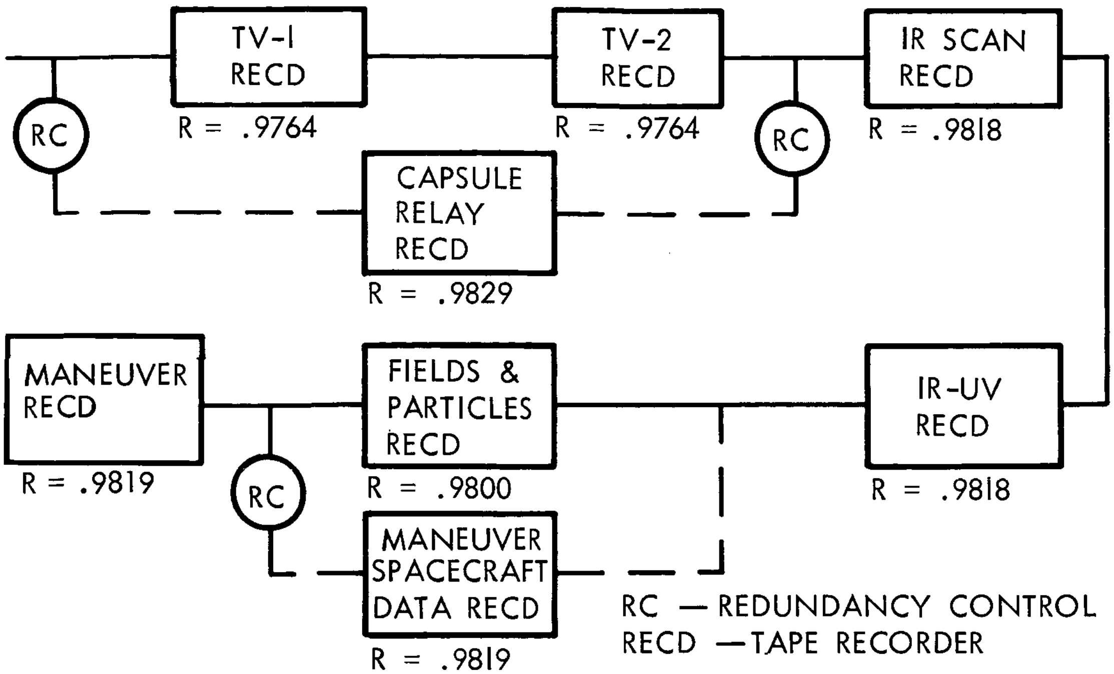
Failure independence in synchronization is maintained by independent count-down circuits off the master sync bus from the power subsystem master oscillator. Loss of master sync is precluded by a multivibrator capable of a free-running mode if the master sync fails. Also, loss of one power-input bus causing complete subsystem failure has been obviated by using dual a.c./d.c. converters which operate off independent power buses.

Results of trade studies performed in Task A are applicable to the preferred design presented here (see Section 4.0 for details). Based on the proposed reliability program disciplines, no parts or materials used in the telemetry subsystem will exhibit life-limiting failure mechanisms for at least three times the total mission period.

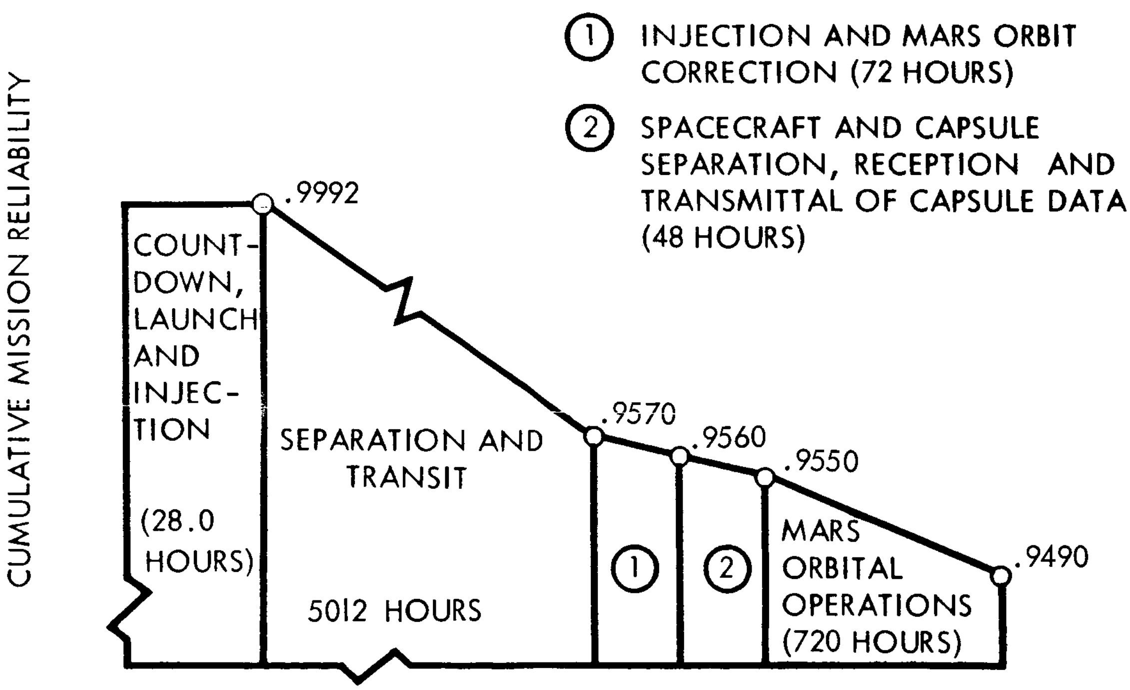
3.17.3.9 Data Storage Subsystem

The data storage subsystem was allocated a reliability of 0.94. The reliability block diagram (Figure 3.17-14) shows that the reliability assessment for the preferred subsystem is 0.9490. Figure 3.17-14 also presents the preferred design's cumulative mission reliability at salient points during the mission.

The reliability of 0.9490 (subsystem design details are given in Section 4.0) was achieved by using functional redundancy with switching capability that provides for backup modes of operations for the



RELIABILITY BLOCK DIAGRAM



MISSION PHASE HAZARD CHART

Figure 3.17-14: Preferred Data Storage Subsystem Reliability Summary

TV and field and particle recorders at critical points during the mission.

The preferred design uses the capsule relay data recorder for backup of both TV recorders after capsule data has been relayed to Earth. This mechanization increases the probability of success of the TV recorders from 0.916 to 0.998. The preferred configuration also significantly improves reliability of the field and particle recorder (from 0.9815 to 0.9997) by utilizing the maneuver recorder as a backup after it has performed its primary function. Because the IR scanner and the IR-UV spectrometer are single thread, a failure in either will result in a substantial loss of capability. However, significant quantities of these data can be real-time transmitted, thereby providing a degraded but useful alternate.

Special efforts have been made to ensure that the data storage subsystem has no life-limiting characteristics that would significantly affect mission success. This effort was directed toward eliminating or nullifying the effects of principal failure modes in the magnetic tape recorders (wearout of belts, tapes, and heads are most critical).

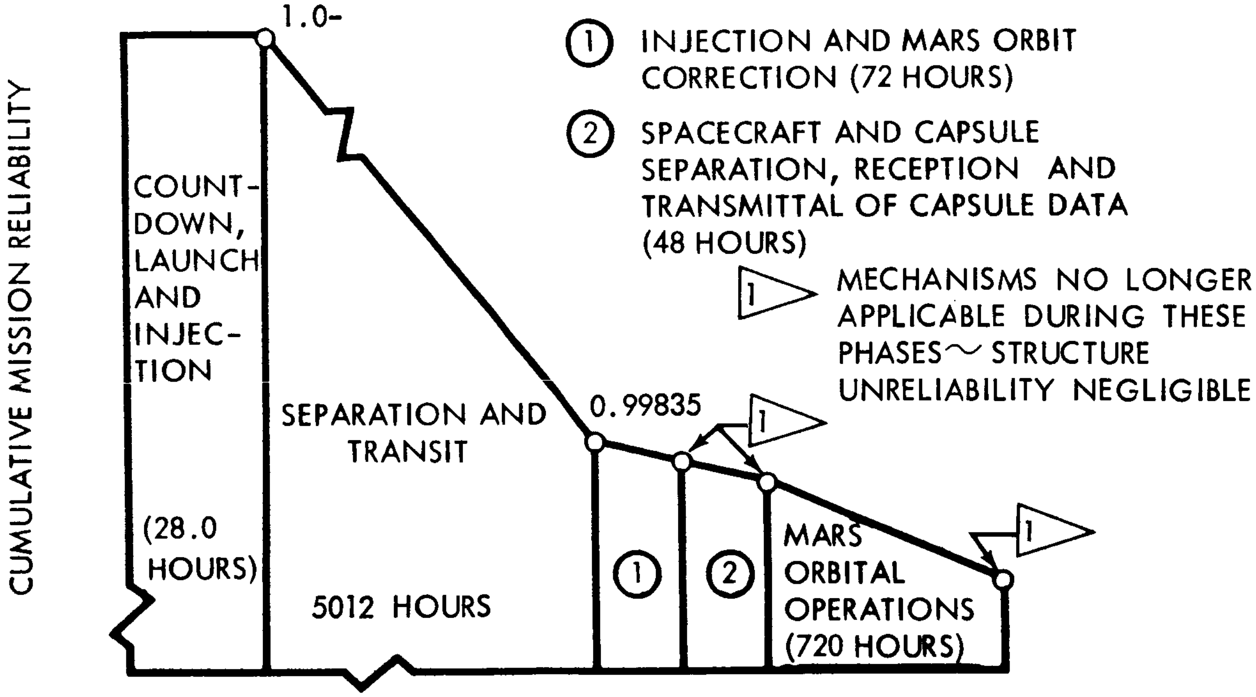
3.17.3.10 Structure and Mechanical Subsystem

Reliability predictions indicate that the 0.998 allocation for this subsystem is attainable. Figure 3.17-15 shows the predicted reliability of the subsystem and its components.

The structure and mechanical subsystem consists of the spacecraft structure and a number of mechanisms not identified as part of other

COMPONENT	MISSION RELIABILITY
SUBSYSTEM ALLOCATION	0.998
STRUCTURE (PRIMARY, EXTERNAL, & PROPULSION)	0.999
HIGH-GAIN ANTENNA RELEASE & DEPLOYMENT MECHANISM	0.99986
LOW-GAIN ANTENNA RELEASE & DEPLOYMENT MECHANISM	0.99988
EMERGENCY FLIGHT CAPSULE SEPARATION MECHANISM	0.99992
GUIDANCE SCAN PLATFORM RELEASE MECHANISM	0.99999
SOLAR PANEL RELEASE MECHANISM	0.99992
PLANETARY VEHICLE SEPARATION MECHANISM	0.99992
MEDIUM GAIN ANTENNA RELEASE & DEPLOY MECHANISMS	0.99986
SUBSYSTEM PREDICTION	0.99835

RELIABILITY OF SUBSYSTEM COMPONENTS



MISSION PHASE SUBSYSTEM HAZARD CHART

Figure 3.17-15: Preferred Structures And Mechanisms Reliability Summary subsystems. Specifically excluded from the subsystem are the mechanisms for solar-panel deployment and high-gain-antenna pointing, which belong to the power and radio subsystems.

Because it is a diverse collection of structures and mechanisms, this subsystem cannot be subjected to customary reliability prediction methods; therefore, the prediction appearing in Figure 3.17-15 was made with the benefit of several simplifying assumptions. It is assumed that in-tolerance performance of all components is essential to mission success. It is also assumed that all components are functionally independent and that no redundancy or backup capability exists. These assumptions are conservative and yield a conservative prediction.

The lack of statistically significant failure date for comparable structures and environments precludes any detailed numerical assessment of subsystem structural reliability. A realistic assessment of the risks involved, using the best available engineering judgment, seems to be the only reasonable approach. Figure 3.17-15 shows an overall structural reliability of 0.999 on this basis. Reducing the failure risk to a minimum involves the following: conservative materials allowables and loads, proven stress-analysis techniques with major attention to points of stress concentration, design for simplicity and easy inspection, design review, stringent quality controls, dynamic environmental testing, failure analysis and corrective action follow-up.

High mechanical reliability can be achieved by using space-proven devices and by thorough environmental testing. Where possible, devices are identical to those used successfully on other space programs. Redundancy is employed wherever feasible. V-band release mechanisms employ redundant explosive nuts, separation mechanisms incorporate redundant ejection springs, and two rotation paths are provided at each hinge joint. Design of V-band release mechanisms will be extremely conservative. Testing to demonstrate design conservatism and functional effectiveness will be accomplished.

Possible subsystem failure modes are listed below. Because various degrees of failure will be associated with many of the listed failure modes, the effects of a specific failure mode may vary from "no effect" to "mission failure," depending on the degree of failure. Therefore, correlation of failure modes and effects has not been attempted.

Possible Failure Modes	Possible Failure Effects
Excessive Yielding	Mission Failure
Excessive Deflection	Secondary Failure
Instability	Degraded Performance
Rupture	No Effect
Inadequate Heat Transfer	
Inadequate Support During Launch-Boost	Mission Failure
Failure to Release for Deployment	Secondary Failure
Incomplete Deployment	Degraded Performance
Failure to Lock in Fully Deployed Position	No Effect
Lock-Indication Failure	
Premature Release or Deployment	
Out-of-Tolerance Performance	

3.17.3.11 Planetary Vehicle Adapter

The reliability allocation for the Planetary Vehicle Adapter was 0.999. Reliability analysis indicates that this goal will be met.

The Planetary Vehicle Adapter supports the Planetary Vehicle prior to separation from the boost vehicle. Static and dynamic loads are transmitted from the Planetary Vehicle, through the adapter, to the nose fairing. The adapter incorporates pyrotechnics, transducers, cabling, coolant ducts, and electrical and coolant in-flight disconnects. The lower adapter will accommodate installation of a destruct mechanism and associated cabling from the S-IVB instrumentation unit.

As with the Planetary Vehicle structure, the assessment of Planetary Vehicle Adapter reliability utilized a considerable amount of engineering judgment. The assumption was made that failure of any major component of the adapter would result in adapter failure. Accordingly, the adapter reliability was assessed as the product of the reliabilities of the structure, the redundant pyrotechnics, the cabling and ducting, the electrical and coolant in-flight disconnects, and the destruct mechanism. For the destruct mechanism, inadvertent functioning was the only failure mode consideration. The probability of this event, based on safety considerations, was set at less than 10^{-6}.

The possible failure modes and effects identified in paragraph 3.17.3.10 for structures are all applicable to the Planetary Vehicle Adapter. Additional failure modes causing minimum loss are: (1) loss of electrical continuity between launch and Planetary Vehicles, (2) inadvertent operation of destruct device, and (3) failure of inflight disconnects to separate.

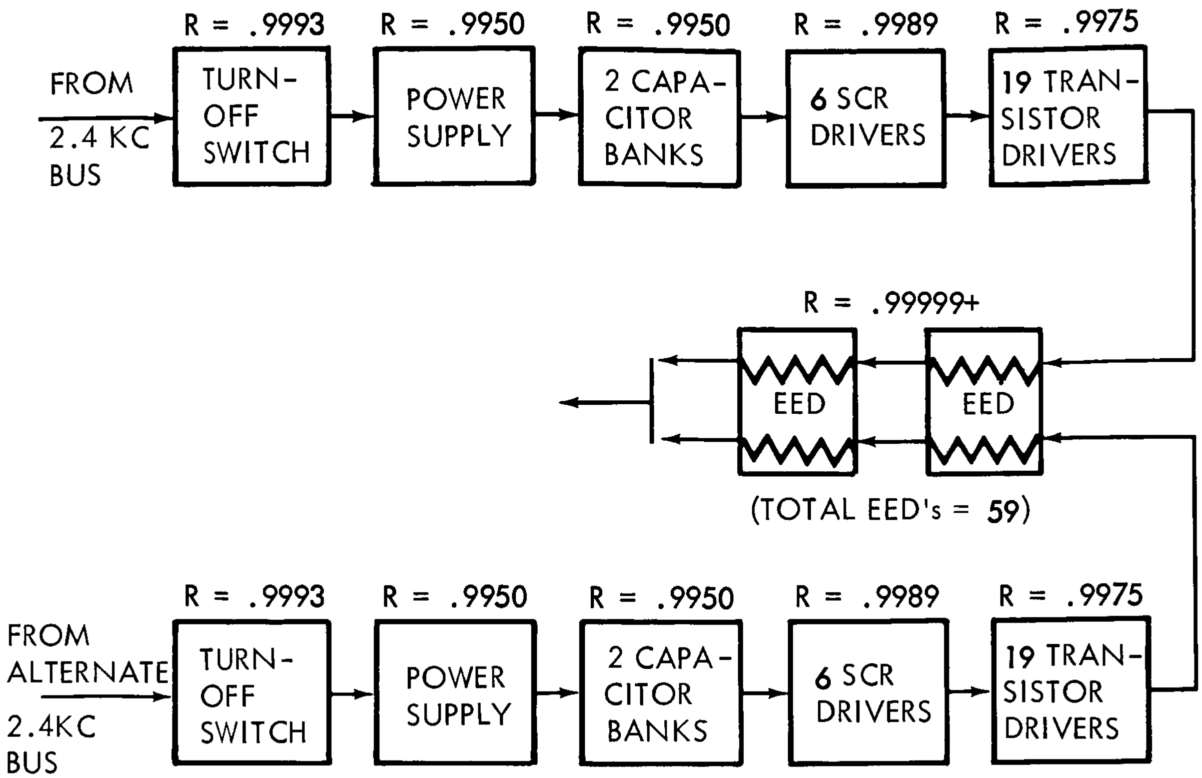
3.17.3.12 Pyrotechnic Subsystem

The reliability allocation of the pyrotechnic subsystem was 0.999. The assessed value of reliability for the mission is 0.99997. A reliability block diagram and mission phase hazard chart for this subsystem is shown in Figure 3.17-16. The block diagram shows the use of complete redundancy where each section of the subsystem takes power from a separate 2.4-kc bus and can fire one bridge wire in each electroexplosive device (EED). Each EED has two bridge wires--either can fire the device. The diagram also shows the mission reliability of the major subsystem components.

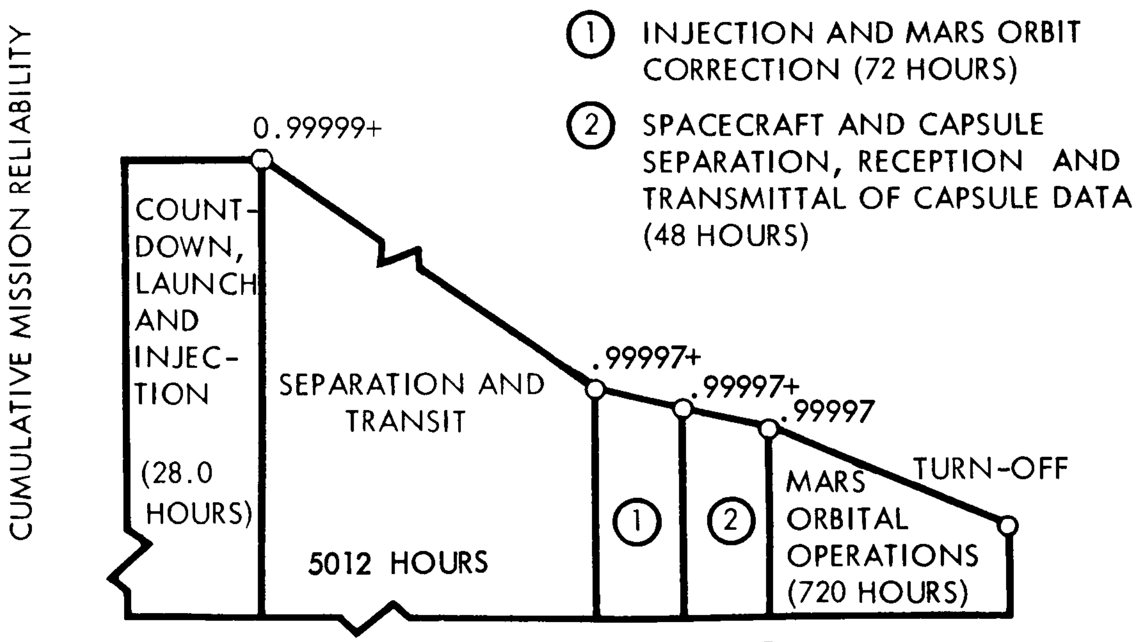
3.17.3.13 Temperature Control Subsystem

The temperature control subsystem uses space-proven louver concepts, insulation, and shielding materials. The assessed reliability for the total mission is 0.999 (Figure 3.17-17), which exceeds the initial allocation of 0.996.

The temperature control subsystem is divided into two sections. One section is the louvers and insulation that control the temperature of the Spacecraft Bus, and the other section consists of the louvers, solar shield, 120-inch shell insulation, and capsule shield that control the temperature of the propulsion area. Heat for spacecraft temperature control is derived from the heat dissipated from electronic equipment and the solar heat that penetrates the insulation and shields. The Spacecraft Bus electronic bays have 13 louver assemblies with 18

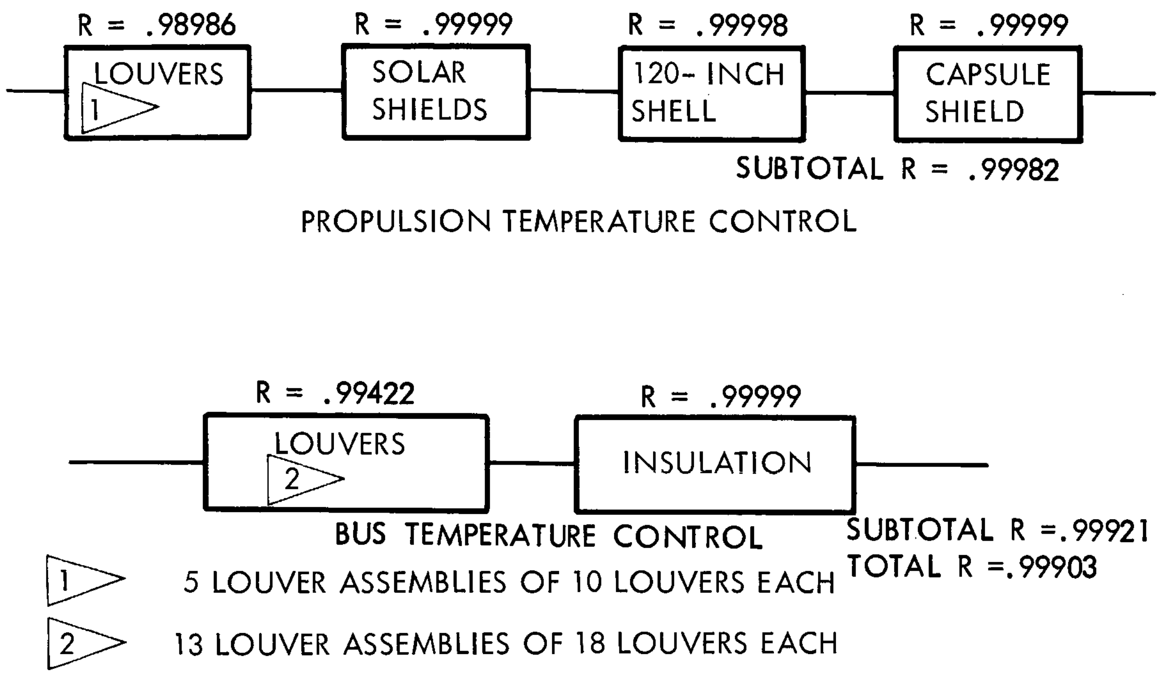


PYROTECHNIC SUBSYSTEM
RELIABILITY BLOCK DIAGRAM

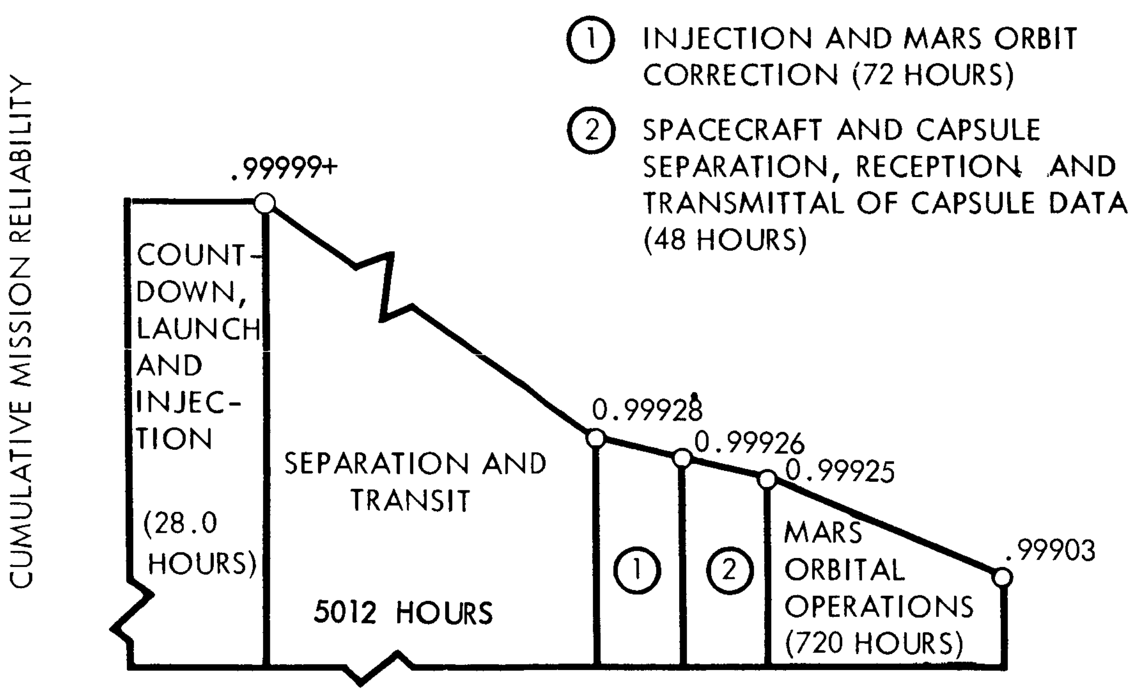


MISSION PHASE HAZARD CHART

Figure 3.17-16: Preferred Pyrotechnic Subsystem Reliability Summary



TEMPERATURE CONTROL SUBSYSTEM
RELIABILITY BLOCK DIAGRAM



MISSION PHASE HAZARD CHART

Figure 3.17-17: Preferred Temperature Control Subsystem Reliability Summary individually actuated louvers. The propulsion temperature control has 5 louver assemblies with 10 individually actuated louvers. If one louver blade fails, it will affect the temperature near the adjacent louvers and they will operate and partially compensate for the failed louver. The louvers are open while in the Earth's atmosphere; they gradually close because of reduced solar-heat absorption as the spacecraft goes toward Mars. This indicates that a failure in the closed position after the spacecraft leaves the Earth's atmosphere will not affect the temperature of the Spacecraft Bus. A failure in the open position after the spacecraft leaves the Earth's atmosphere is compensated for by louver design--several louvers may fail as long as there are four good louvers between failed louvers. More failures than this will lower the temperature, which may degrade the spacecraft's operations. There are no known life-limiting characteristics associated with the louvers. Wear is not a problem because the number of louver operations will be low.

Multilayer insulation can lose some of its layers or can be penetrated by meteoroids without degrading performance to the point where it will affect system performance. The insulation has no known life-limiting characteristics.

In addition to the temperature control system described above, a network of heaters, activated by ground command, can compensate for any equipment that is off. This also provides temperature protection for the remaining equipment in the event of any equipment malfunction or loss of heat in parts of the spacecraft.

3.17.3.14 Cabling Subsystem

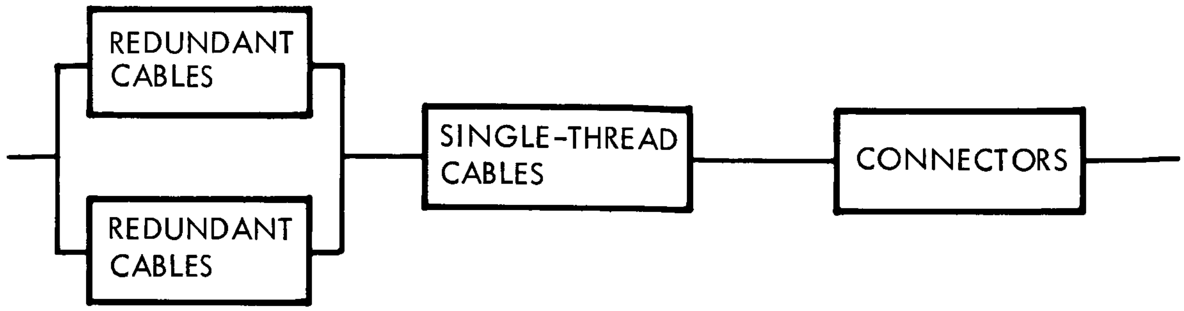
Reliability of the cabling subsystem (Figure 3.17-18) was assessed at 0.996. The critical circuits use redundant wires so that a single pin or wire failure will not prevent mission success.

The cabling consists of approximately 5000 wires and 300 connectors. It was assumed that 15 percent of the signal wires are redundant and all the pyrotechnic circuits are redundant. After a particular pyrotechnic circuit is fired, it was eliminated from the calculations.

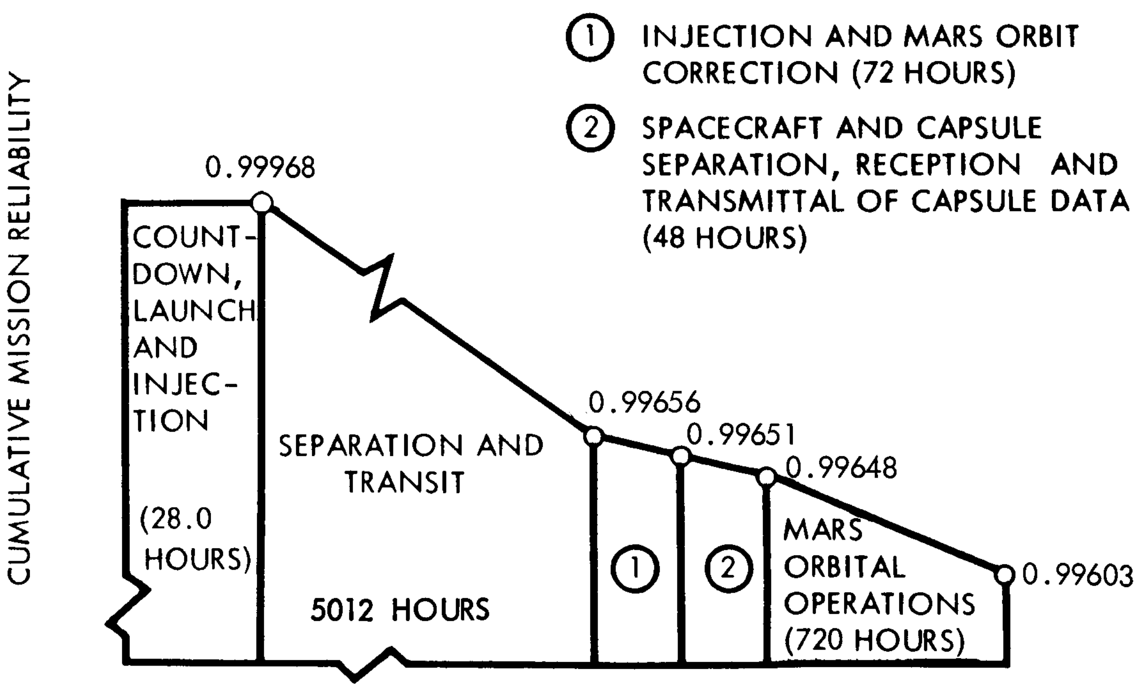
Further definition of the cabling subsystem is required to determine if the reliability allocation of 0.999 is compatible with the overall assessment. Connectors and cables will be selected to meet the space environment. The connectors will use crimp-type pins. All cables or wire bundles will be adequately clamped to withstand vibration. Electronic cabling to deployed mechanisms will be protected against micrometeoroid damage. There are no known critical life-limiting characteristics associated with this subsystem.

3.17.3.15 Propulsion System

The evaluated reliability of the propulsion system is 0.9960 compared to an initial allocation of 0.995. The probability of any malfunction resulting in catastrophic failure has been minimized by design modifications to the Task A subsystem, which include: (1) elimination of the need to sense malfunctions and switching, (2) designing configurations so that the only components whose individual malfunctions



CABLING SUBSYSTEM RELIABILITY BLOCK DIAGRAM



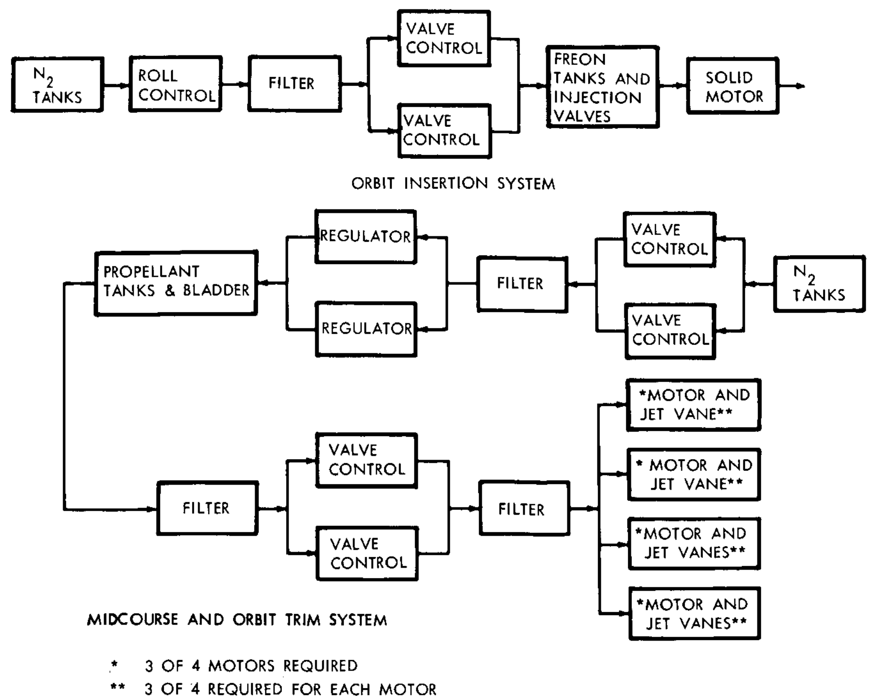
MISSION PHASE HAZARD CHART

Figure 3.17-18: Preferred Cabling Subsystem Reliability Summary would result in catastrophic failure have negligibly small probabilities of failure, and (3) increasing emphasis on alternate paths by which functions may be successfully completed.

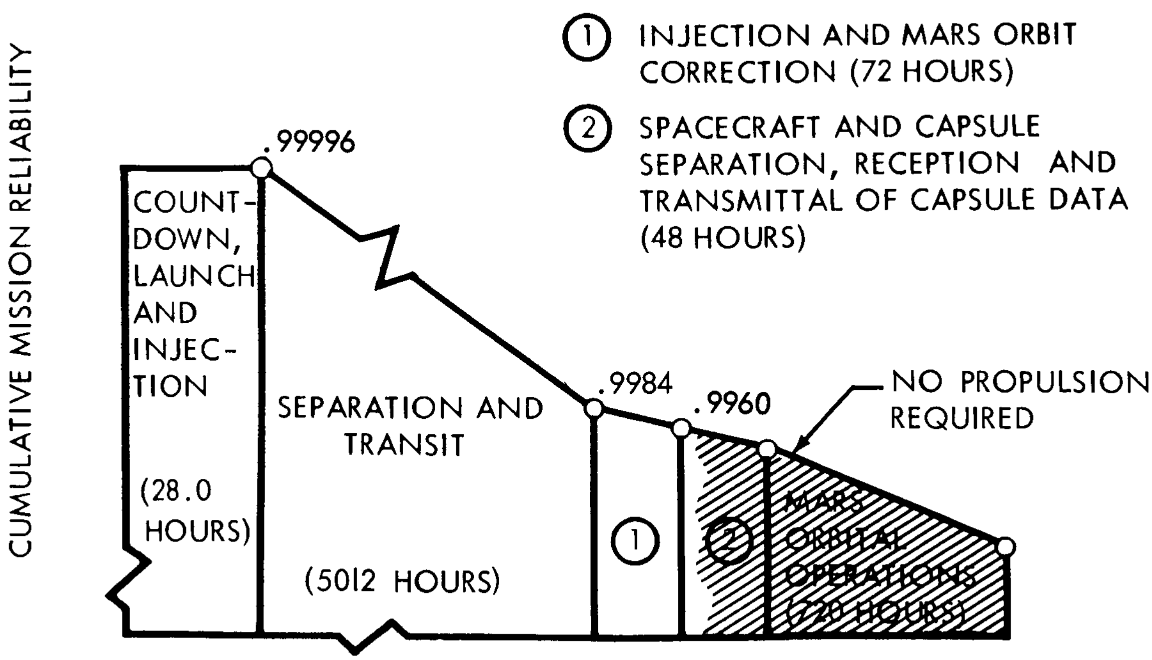
The hazard chart of system reliability versus mission time is shown in Figure 3.17-19.

The system provides thrust for trajectory correction maneuvers, orbit insertion, and orbit trims. Four monopropellant liquid-rocket engines perform two midcourse corrections and two orbit-trim maneuvers. A single solid-propellant rocket motor performs the orbit-insertion maneuver. All engines have thrust vector control (TVC) capability. For a detailed description see Volume A, Section 4.3.

Alternate paths of operation have been provided and can accomplish required functions. The block diagram in Figure 3.17-19 shows the alternate paths and redundancy features. In the monopropellant engine, pressurizing gas is delivered from the supply to the propellant tanks through valve legs redundant for the fail-closed mode. Fail-open and leakage backup is provided by quad pressure regulators. Propellant valving from the tanks to the engines is redundant for the fail-open, fail-closed, and leakage modes. Engine redundancy is provided by the capability to perform course corrections with any three of the four engines. TVC can be accomplished with any three of four independently controlled jet vanes on each engine. Orbit-insertion TVC for the solid-propellant motor is designed with redundancy in pressurization valves, pressure regulation, and roll-control valves. Components of the propulsion system that depend on low failure probability rather than



RELIABILITY BLOCK DIAGRAMS — PROPULSION SUBSYSTEM



MISSION PHASE HAZARD CHART

Figure 3.17-19: Preferred Propulsion Subsystem Reliability Summary redundancy to function successfully include: tanks for pressurizing gas storage, piping, filters, propellant tanks, propellant expulsion bladders, and the solid-propellant rocket engine.

The operating life of the preferred propulsion system is not limited by wearout or fatigue of operating components. Depletion of stored propellant or pressurizing gas because of excessive leakage during the nominal mission would prevent completion of the propulsion subsystem function. The probability of this occurence has been accounted for in the analyses.

Reliability evaluations were conducted for the three candidate propulsion system configurations: (1) modified Minuteman/monopropellant, (2) Titan III-transtage/monopropellant, and (3) LEM-descent/monopropellant.

Similar components in all the configurations were assigned the same failure rate in the analyses. The reliability figures represent the potential probability of success of the configuration types. They do not reflect the test experience actually obtained for the transtage or Minuteman engines. Results of these evaluations are shown on a component comparison basis in Table 3.17-6 (details are given in Volume C).

Table 3.17-7 summarizes the single-effect failure modes of the preferred propulsion configuration. Listed single-effect failure modes are the ones that cause some degradation in mission success or cause loss of mission. All functions performed by equipment backed up by redundant performance capabilities have been eliminated from these summaries, with the exception of two listed on the bottom of Table 3.17-7. These two are noted as marginal single-failure effects.

Table 3.17-6: PROPULSION SYSTEM RELIABILITY TRADE STUDY ASSESSMENTS

	Solid Plus Mono MC ($n \lambda t \times 10^6$)	LEM Plus Mono Settling ($n \lambda t \times 10^6$)	Transtage Plus Mono Settling ($n \lambda t \times 10^6$)
Tank, Pressurization	855	855	855
Valves, Pressurization	6	28	12
Filter, Pressurization	1	3	2
Regulator, Press.	Negligible	23	Negligible
Accumulator	N/A*	---	---
Check Valve	Negligible	1	Negligible
Relief Valve	10	17	15
Tank, Prop.	896	1710	821
Filter, Prop.	2	6	---
Valves, Prop.	Negligible	---	260
Engine, Liquid	Negligible	1350	2000
TVC	Negligible	Negligible	2
Engine, Solid Engine	50	N/A	N/A
TVC, Solid Engine	2227	N/A	N/A
Settling Mono System	N/A	5653	5368
$n \lambda t \times 10^6$	4047	9646	9335

* N/A--Not Applicable

TABLE 3.17-7 FAILURE MODE ANALYSIS - SINGLE FAILURE EFFECT
PROPULSION SUBSYSTEM

Item	Function	Failure Mode	Relative Chance of Occurrence	Effect	Remedy-Alternate, Etc.
General- All Piping, Piped-in Components & Tanks *	Contain gas or fluid	Major leak to external surroundings	Remote	Mission Loss	Structural design Control over Mfg. processes should reduce this to an extremely rare event
Filter	Filter contaminants from regulators	1) Fail to filter (open)	Remote	Degrade performance midcourse correction capability lost	
		2) Clogged	Possible		Adequate oversize can nearly eliminate
Diaphram	Pressurize fuel without contamination from N_2	1) Leak or structural fracture	Possible	Contaminate fuel, fuel locks & stoppage. Engine out -- degrade mission	Manufacture and Quality Control over good design will minimize
Fuel Filter	Filter contaminants from fuel to prevent contamination getting to solenoid control valves	1) Fail to filter (open)	Remote	Increase probability of engine stop failure.	Checkout procedure should eliminate
		2) Clogged	Possible	Possible degradation of midcourse correction control Midcourse correction capability	Adequate oversize can nearly eliminate
Jet Vane Assembly	Deflect engine gases to affect moment control	1) Jammed in an extreme control position	Possible	One engine thrust degraded slightly No effect	Can stand one engine out
Rocket Engine	Provide thrust, 3 out of 4 must work	1) Fail to start 2) Shuts down early		No effect	One engine out allowed

* Common to all Liquid Systems

3.17.3.16 Spacecraft Operational Support Equipment (OSE)

The mission success criteria for the Voyager Project, and the definition of mission phases, dictate that reliability of the OSE be established in two categories related to two different operational phases, namely: (a) a reliability requirement that will be a measure of the probability of reaching and maintaining a state of readiness to launch (prelaunch reliability), and (b) a reliability requirement associated with the equipment used during the mission (starting with the terminal countdown or mission reliability).

Qualitative Reliability Consideration--Appropriate parts of the qualitative requirements summarized in Section 3.17.1.1 are levied on both categories of OSE, with emphasis on protection from overstress or degradation of the spacecraft. Design compliance with these requirements will be evaluated by the methods described in Section 3.17.2.2.

Category 1, Prelaunch Reliability--Generally, the OSE of this category includes the equipment required for assembly, handling, shipping, servicing, testing, and checkout of spacecraft subsystems. Numerical reliability requirements to be associated with this equipment and prelaunch operations have not been established. These requirements will be greatly influenced by such factors as the philosophy advanced for preventive and corrective maintenance, work-around capability, time and equipment availability during critical prelaunch periods, and definition of critical prelaunch periods with attendant OSE operational requirements.

Category 2, Mission Reliability--A reliability goal of 0.99 has been allocated to this category of equipment. For this purpose, reliability has been defined as the probability that the OSE will successfully perform its required function throughout the mission, starting with the terminal countdown of the launch and transit phase of the mission. For compatibility with overall mission success relationships, successful performance is considered to be:

1) Performance of the required functions without an OSE failure that causes a failure in the spacecraft;
2) Performance of required functions without failure of OSE to detect those spacecraft malfunctions for which the OSE is designed to detect;
3) Performance of required functions without a failure that would cause the mission to be rescheduled once the terminal countdown has been started.

The foregoing reliability estimate is believed to be consistent with reliability experience with equipment having similar operational environments and functional and parts complexity. Reliability evaluations and allocations to specific equipment items to be performed later in the program will be influenced by consideration of qualitative requirements and design constraints. It is intended that this equipment category will be subjected to appropriate reliability disciplines comparable to those imposed on the spacecraft equipment.

3.17.3.17 Science Payload Subsystem

The science payload, which is GFE, is used to gather scientific data during the interplanetary transit phase and while orbiting Mars.

Because the science payload is GFE and the reliability is controlled by GFE specifications, the evaluations are used only to determine the overall effect on mission reliability. A total of 15 experiments are to be conducted; of these, seven will be conducted during the Mars orbit only and are considered for this reliability assessment to be primary to mission success. These seven planetary experiments are listed in Group IV of Table 3.17-8 and have an assessed reliability of 0.8001. Four of the seven experiments will operate only during a portion of each Mars orbit.

When the nine interplanetary experiments are included, as listed in Group V of Table 3.17-8, the total assessment becomes 0.5031. Groups I through III of the table show the reliability for other experiments. All 16 experiments operate independently so that a failure of any single experiment will not jeopardize the other experiments. Two units are common to all the science experiments: the data automation equipment, which is used to sequence and condition the data; and the science power switching, which controls power to all the experiments. A failure in science power switching may affect only one experiment because the power to each experiment is controlled by separate relays and relay drivers. The failure rate for the science power switching, which was used in Groups I through IV of Table 3.17-8, was adjusted to compensate for this independence. The data automation equipment processes data in both real and non-real time. A failure in one mode of collecting data will not affect the other mode. Also, the data automation equipment has sections that work with one specific experiment. These specific sections may fail without affecting the

TABLE 3.17-8: SCIENCE PAYLOAD RELIABILITY RANGE

		λt	
I	Infrared Scanner Only		
	Data Automation Equipment	0.0653	
	Infrared Scanner	0.0068	
	Science Power Switching (15 percent used)	0.0044	
	Total	0.0765	0.9264
II	Photoimaging Device Only (T.V.)		
	Data Automation Equipment	0.0653	
	Photoimaging Device (one experiment only)	0.0294	
	Science Power Switching (15 percent used)	0.0044	
	Total	0.0991	0.9057
III	Infrared Scanner & Photoimaging Devices		
	Data Automation Equipment	0.0653	
	Infrared Scanner	0.0068	
	Photoimaging Devices (two experiments)	0.0589	
	Science Power Switching (45 percent used)	0.0132	
	Total	0.1442	0.8657
IV	Planetary Only		
	Data Automation Equipment	0.0653	
	Infrared Scanner	0.0068	
	Photoimaging Devices (two experiments)	0.0589	
	IR Spectrometer	0.0194	
	UV Spectrometer	0.0407	
	RF - Noise Detector	0.0045	
	Ionosphere Sounder	0.0063	
	Science Power Switching (73 percent used)	0.0211	
	Total	0.2230	0.8001
V	All Experiments		
	Data Automation Equipment	0.0653	
	Infrared Scanner	0.0068	
	Photoimagining Devices (two experiments)	0.0589	
	IR Spectrometer	0.0194	
	UV Spectrometer	0.0407	
	Bistatic Radar	0.0426	
	RF - Noise Detector	0.0045	
	Ionosphere Sounder	0.0063	
	Cosmic-Ray Telescope	0.0308	
	Magnetometer	0.0833	
	Plasma Probe	0.0736	
	Trapped-Radiation Detector	0.0346	
	Cosmic-Dust Detector	0.0121	
	Ion Chamber	0.0691	
	Gamma Ray Spectrometer	0.0596	
	Gravimeter	0.0501	
	Science Power Switching (100 percent used)	0.0293	
	Total	0.6870	0.5031

other sections of the data automation equipment. Electrical power for these experiments is furnished through redundant power buses. No life-limiting characteristics are anticipated at this time.

3.17.3.18 Launch Vehicle

Although the launch vehicle is not primary to Boeing's Task B effort, its reliability was assessed to provide overall-system visibility. The launch vehicle consisting of S-IC, S-II, and S-IVB stages, an instrument unit, and a nose fairing must inject the two Planetary Vehicles into acceptable interplanetary transfer trajectories. A probability of success of 0.90 was allocated for this function. Assessment of the 0.90 was performed by analyzing each stage.

Chart A of Figure 3.17-20 shows reliability assessment, derived from the design assessment, for the Saturn V stages. Chart B shows current estimates (based on latest firing data) for the Atlas-Mercury and composite Mercury-Gemini shots. These data provide typical booster maturity points. Chart C represents a typical booster-reliability-growth curve showing the cumulative reliability for 33 firings of the Titan II booster. By combining the data of Chart A (as a starting point), the typical growth of Chart C, and the maturity range of Chart B, the Voyager 1971 launch-vehicle reliability was forecast as shown in Chart D. This analysis indicates that an objective of 0.90 for the Voyager launch vehicle is feasible. This conclusion is further supported by the experience of the Saturn I and manned space shots, where conservative design, component screening, and rigorous control achieved significant gains in booster reliability.

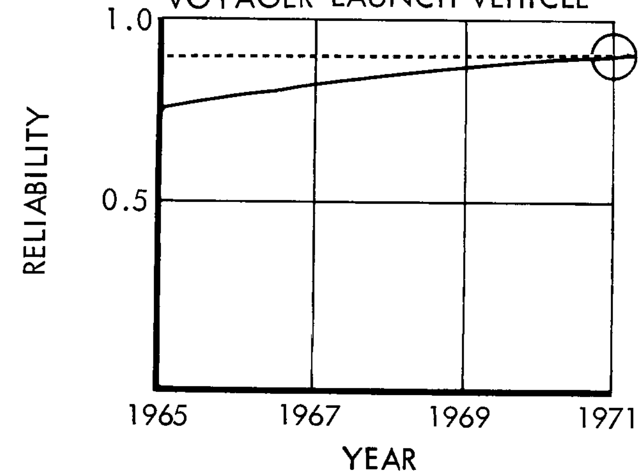
A BOOSTER STAGE

BOOSTER STAGE	CURRENT ASSESSMENT
S-IC	0.99
S-II	0.89
S-IVB	0.91
INSTRUMENT UNIT	0.99
TOTAL	0.79

B OTHER LAUNCH VEHICLES (MATURE)

ATLAS (MERCURY SHOTS)	0.90
MERCURY-GEMINI SERIES (REDSTONE, ATLAS, TITAN II)	0.86

D 1971 FORECAST VOYAGER LAUNCH VEHICLE



C TITAN II GROWTH

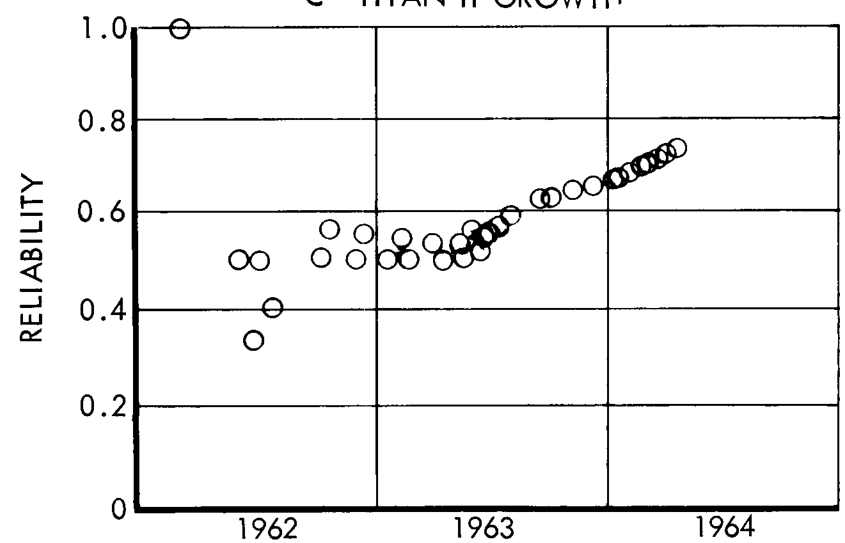


Figure 3.17-20: Launch Vehicle Reliability

3.17.3.19 Mission Operations System (MOS) and Tracking and Data System (TDS)

Although these systems are not primary to Boeing's Task B reliability effort and have not been evaluated in detail, the reliability for the combination of the MOS (including the mission-dependent equipment) and the TDS is expected to be such that their combined probability of success (P_S) in support of the Voyager mission is 0.98.